NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA PROGRAM APOLLO WORKING PAPER NO. 1339

# TECHNIQUES AND RESULTS OF APOLLO 4 SERVICE PROPULSION SYSTEM POSTFLIGHT ANALYSIS





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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
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PREPARED BY


John D. Norris

John D. Norris
AST, Systems Analysis Section

F. Don Freeburn

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AST, Systems Analysis Section


AUTHORIZED FOR DISTRIBUTION

Warren Gillespie, Jr

for Maxime A. Faget
Director of Engineering and Development


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# TECHNIQUE AND RESULTS OF APOLLO 4 SERVICE PROPULSION SYSTEMS POSTFLIGHT ANALYSIS

By John D. Norris and F. Don Freeburn

## SUMMARY

The performance of the Spacecraft 017 service propulsion subsystem during the Apollo 4 mission was analyzed using the Apollo Propulsion Analysis Program. A satisfactory correlation of the flight data was achieved only for the portion of the burn after crossover (storage tank depletion). The resulting engine flight performance corrected to standard inlet conditions resulted in the following values:

1. Thrust — 21,503 pounds

2. Specific impulse — 312.06 seconds

3. Mixture ratio — 2.007

These values are 0.01 percent higher, 0.27 percent lower, and 0.37 percent higher, respectively, than the acceptance test log results (also reported at standard inlet conditions). The uncertainty ($3\sigma$) of the above analysis results is estimated to be 0.50 percent for thrust and specific impulse and 1.8 percent for mixture ratio. The uncertainty of the acceptance test values is even greater thus the agreement is well within the expected tolerances.

The Apollo 4 service propulsion system operation was nominal, and the two detailed test objectives, no-ullage start and long-duration burn, were accomplished to a satisfactory degree.

## INTRODUCTION

### Analysis Description

The Apollo unmanned flight tests are necessary for full qualification of the propulsion systems. If adequately instrumented and properly analyzed, these Apollo flights afford the best opportunity to determine accurately the integrated engine and system propulsion performance under the actual environmental conditions. If this can be accomplished, it will provide a greater level of confidence and allow for smaller uncertainties in planning later flights.

In order to accurately determine steady-state propulsion performance from flight data, TRW has developed the Apollo Propulsion Analysis Program (APAP). The approach employed for the performance analysis program is a minimum variance estimation technique which correlates flight data with ground test results. Applicable flight data which pertain to propulsion system performance are utilized. An important feature of the technique is that the accuracy with which the parameters are determined is also provided. The computer program which embodies this technique is general and is used for all three Apollo primary propulsion systems; the service propulsion subsystem (SPS), descent propulsion subsystem, and the ascent propulsion subsystem.

A propulsion flight analysis program that was used extensively on the Thor, Atlas, Titan, and Minuteman ballistic missile programs had previously been developed by TRW. The present Apollo flight analysis program is a refined, more versatile extension of the original ballistic missile version. Apollo 4 is the first flight where the analysis was performed using the APAP. The analysis of previous SPS flights (Spacecraft 009 and 011) used the simpler and more specialized original version.

## Service Propulsion Subsystem Mission Description

The Apollo 4 mission included the third flight test of the SPS. The primary SPS test objectives were to demonstrate a satisfactory start without a reaction control subsystem (RCS) settling maneuver and to determine SPS performance during a long-duration burn. The Apollo 4 mission plan called for two SPS burns; a short burn of approximately 16 seconds duration followed by a long burn of approximately 271 seconds duration. The no-ullage start was to be demonstrated on the first burn, and the long burn performance determination objective was to be satisfied by the second burn.

The first SPS burn lasted 16 seconds, with cut-off by guidance and navigation (G&N) command. Following the first SPS burn, the spacecraft was alined to a specific attitude to achieve a thermal gradient across the command module heat shield. Approximately 4-1/2 hours later, the spacecraft was reoriented to the long burn ignition attitude. A service module (SM) RCS ullage maneuver was initiated 30 seconds prior to SPS ignition. The second burn ignition was initiated by G&N command at 08:10:54.8, range time. Subsequently, a redundant thrust-on was commanded from the ground, which disenabled the SPS thrust-on/off command of the G&N subsystem. The SPS thrust-on ground command required that the SPS thrust-off also be commanded from the ground. The second SPS thrust-off was initiated by ground command at 08:15:35.4. The burn duration was 10.1 seconds longer than planned, resulting in a higher than planned velocity gain.

## Subsystem Description

The SPS consists of three primary subassemblies: (1) engine system, (2) propellant storage and feed system, and (3) pressurization system. A functional flow diagram is shown in figure 1.

The engine system produces a nominal thrust of 21 500 pounds, operating at a nominal mixture ratio of 2.0. The combustion chamber is ablatively cooled. The propellants are earth storable and hypergolic. The fuel (A-50) is a 50/50 blend (by weight) of unsymmetrical dimethylhydrazine and anhydrous hydrazine; the oxidizer is nitrogen tetroxide ($N_2O_4$). The engine bipropellant shutoff valve is pneumatically actuated by gaseous nitrogen. Bolted to the engine chamber is a nozzle extension composed of two columbium sections to an area ratio of 40:1 and a titanium section to the exit (62.5:1). Nozzle extension cooling is by radiant heat transfer to space.

Fuel and oxidizer are each contained in a set of two cyclindrical tanks connected in series. The downstream tanks are called the sump tanks and are directly connected to upstream storage tanks by crossover lines and standpipes. Each sump tank outlet contains propellant retention screens and a propellant retention reservoir which retain propellant over the propellant feedline inlet during near zero-g conditions and reduce the propellant settling time requirements. Thrust from the SM RCS engines provides for propellant settling in addition to that maintained by the above mentioned retention devices.

The helium pressurization supply is contained in two spherical pressure vessels at an initial nominal pressure of 4000 psia and ambient temperature and is isolated from the fuel and oxidizer tanks during engine shutoff by two continuous-duty-operated solenoid valves. Two dual-stage regulators, arranged in parallel, are located downstream of the solenoid valves and provide pressure-regulated helium to the fuel and oxidizer tanks. Two sets of check valve assemblies, arranged in series-parallel configurations, prevent fuel or oxidizer from entering the pressurization system. Pressure relief valves prevent overpressure in the propellant tanks. Heat exchangers are used in the helium lines to condition the helium to a temperature approximately that of the propellant in the tanks.

The following are the major SPS hardware and flight configuration differences from the two previous missions on which the SPS has flown:

1. The propellant storage tanks were partially loaded. Previous flights flew with only propellants in the sump tanks. The partial load in the storage tank made possible the longest duration SPS flight burn to date ($\sim$281 seconds). During this burn, the effects of storage

tank depletion (propellant crossover) on SPS performance could be assessed for the first time.

2. Propellant retention screens were installed in the propellant sump tank bottoms which, in addition to the propellant retention reservoirs located there, permitted the first demonstration of a SPS no-ullage start.

3. The flight combustion stability monitor (FCSM) was flown activated (previously inactive) during the two SPS burns, though it was made inoperative into the second burn when the G&N mode of operation was overridden by the backup ground command.

4. The gaging system was in the primary mode during the flight, telemetering only primary gaging data during each SPS burn. This permitted the first opportunity to obtain flight data from the primary gaging system in the propellant storage tanks.

Acknowledgments

This report was not the result of an individual effort, but rather the result of an integrated NASA Manned Spacecraft Center (MSC), TRW, and Lockheed Electronics Company (LEC) support effort. The authors wish to express their appreciation to the following organizations and personnel who made significant contributions to the AS-501 SPS analysis: TRW Systems, who developed the analysis capabilities and assisted in the analysis and report writing; Mr. Joe Fries, NASA, who had primary responsibility for the data processing and detailed analysis; Mr. Richard Rosencranz, NASA, for processing the acceleration data; and Mr. John Moorhead and Mr. Donald Landry, LEC, for data processing and plotting.

## SERVICE PROPULSION SUBSYSTEM OPERATION

The primary propulsion system operated nominally during the first 16-second burn. Data plots of the tank, engine inlet, and chamber pressures are shown in figures 2 through 4 for the first burn. The second burn also indicated nominal operation with the pressures reading approximately the same as for the first burn and holding steady until crossover (storage tank depletion) which occurred about 80 and 81 seconds into the second burn for oxidizer and fuel, respectively. During crossover, the engine inlet and chamber pressures rose as expected and were then steady through the remainder of the burn. Data plots of the tank, engine inlet, and chamber pressures are shown in figures 5 through 7 for the second burn. In addition, plots of data from the gaging system are shown in figures 8 and 9 for the second burn.

# STEADY-STATE ANALYSIS PROCEDURES

## Analysis Technique

The major effort for this report was concentrated on determining the SPS performance during the Apollo 4 mission. This was accomplished by utilizing the APAP. The program utilizes a minimum variance estimation technique in conjunction with pertinent data from the flight and from previous static tests, in addition to the physical laws which describe the behavior of the propulsion/propellant systems and their interactions with the spacecraft. The program embodies error models for the various flight and static test data that are used as inputs, and by iteration methods, arrives at estimations of the system performance history, initial propellant weights, and spacecraft weight which "best" (minimum variance sense) reconcile the available data. The technique is to determine the coefficients of the propulsion and propellant systems performance parameters in the error model that minimizes the quantity $X^2$.

$$X^2 = \sum_{i=1}^{m} \sum_{j=1}^{n} \frac{\left(Z_{ij}^* - Z_{ij}\right)^2}{\sigma_{ij}^2}$$

where $X^2$ = a function to be minimized

$Z_{ij}^*$ = a flight test data point

$Z_{ij}$ = value corresponding to the flight test data calculated by the simulation model

$\sigma_{ij}$ = a priori estimate of the standard deviation of the data point

$m$ = the number of data measurements used

$n$ = the number of data points per measurement

The key to a successful postflight analysis is the extremely accurate thrust acceleration that can be calculated from the Apollo guidance computer (AGC) $\Delta V$ data. Assuming that there were no unknown biases present, it is estimated that the acceleration during the flight was determined within $\pm 0.02$ ft/sec$^2$. This would result in an accuracy of approximately 0.10 percent, which is more than an order of magnitude better than any other propulsion measurements. From the acceleration data, the time history of the ratio of thrust to weight can be determined. Fitting this ratio with the other sources of information previously mentioned and adjusting the initial conditions and measurements according to their estimated sigmas in an iterative procedure results in a converged condition which represents the best estimate of the true state.

The program results presented in this report were based on simulations using data from the flight measurements listed in table I.

## Data Sources

Propellant loaded.- The oxidizer storage tank was filled by overflowing the sump tank through the crossover line at a flow rate of approximately 60 gal/min. The trapped sump helium ullage gas was being entrained by the oxidizer that was being transferred and resulted in a sump ullage pressure decrease with a corresponding propellant level rise of about 3 inches above the standpipe in the sump tank. As a consequence, more oxidizer was loaded than planned. Based on the analysis of the storage tank primary gaging system, the sump tank propellant level was raised at pressurization approximately 6 inches above the standpipe. This level was above the maximum gageable level of the sump primary gaging probe. The oxidizer sump ullage volume was reduced by approximately 2 cubic feet.

The fuel storage tank servicing rate was reduced from 60 to 15 gal/min. This alleviated the helium entrainment problem on the fuel side. The resulting propellant loading was as follows.

| Tank | Oxidizer, lb | | Fuel, lb | |
| --- | --- | --- | --- | --- |
| | Planned | Actual | Planned | Actual |
| Storage tank[a] | - | 4 643 | - | 2 278 |
| Sump tank[a] | - | 15 549 | - | 7 795 |
| Total | 20 077 | 20 192 | 10 022 | 10 073 |

[a]Includes gageable and nongageable quantities.

Propellant densities.- Eight $N_2O_4$ density samples at 39.2° F were measured from propellants used to service the SPS and RCS. It was not known which samples were for SPS and which samples were for RCS; however, since there was very little variance between the samples, the average value was used. The mean for $N_2O_4$ was 1.4830 g/ml (92.581 $lbm/ft^3$) at 39.2° F. Three A-50 density samples for the SPS were analyzed. The mean for A-50 was 0.90047 g/ml (56.215 $lbm/ft^3$) at 77° F. The standard density equations as a function of temperature were shifted to reflect the measured density value. The resulting equations were

$$\rho_{ox} = 95.64 - 0.078035(T) + 0.000699(P - 14.7)$$

$$\rho_{fu} = 58.66 - 0.031838(T) + 0.000368(P - 14.7)$$

The problem of determining the flight densities for the Apollo 4 SPS analysis is magnified by the absence of all direct propellant temperature measurements from the operational telemetry (TM) list.

The ambient air temperatures for 12 hours prior to launch were obtained from Kennedy Space Center (KSC), and were 67° F. If the propellant temperatures are assumed equal to the mean ambient air temperature, both the oxidizer and fuel temperature would be approximately 67° F. Tape recorder data recorded during boost for measurements SP2075, SP2076, SP2077, SP2078, which measure helium temperature in and out of the oxidizer and fuel heat exchangers, respectively, were indicating approximately 68° F for oxidizer and 72° F for fuel. During boost there is no helium flow, and it is reasonable to assume these measurements should read approximately the temperature of the propellant in the respective heat exchanger. These, however, are local temperatures and may not be representative of the bulk temperatures. The helium bottle line outlet temperature indicated a reading of about 85° F. This value is somewhat in conflict with the other available measurements. There were to be two Structures and Mechanics Division temperature measurements on the fuel propellant tanks; however, these measurements were moved. It was planned that these measurements would be the prime source of information.

After considering the above data, it was decided to assume a temperature of 70° F for both propellants for this report. Using 70° F and the nominal interface pressures, the following densities were calculated from the density equations and used for this analysis

$$\rho_o = 90.28 \text{ lbm/ft}^3$$

$$\rho_f = 56.49 \text{ lbm/ft}^3$$

It should be emphasized that propellant densities strongly affect the performance analysis, and the absence of direct propellant temperature measurements compromises the estimates of the densities thereby decreasing the confidence in the analysis results.

## Data Reduction

Upon receipt of the AS-501 station data tapes, the Data Reduction Center processed the tapes and produced a phase I tape. The phase I tape is band passed, wild point edited, quantified, and packed onto a Univac 1108 compatible binary tape. Also stripped off the phase I tape is the guidance computer word which is put out on a separate tape in the standard AGC down-link list format. These tapes are supplied to the Primary Propulsion Branch by the Computation and Analysis Division.

The Univac 1108 binary tape is then processed through a decommutation program which produces a raw data tape. The raw data were smoothed using an orthogonal polynominal sliding arc filter and sliced at a sample rate suitable for input to the analysis program. The raw data and smoothed data were plotted on Calcomp plotters.

The guidance computer data are also specially processed. The data, which are in the form of velocity increment counts, are first edited to eliminate bad data, and then the data are scaled, biased, smoothed, sliced, and converted to acceleration. The acceleration data are merged with the smoothed propulsion system data. This resultant tape is the input tape to the analysis program.

# STEADY-STATE PERFORMANCE

## Analysis Program Results and Critique of Analysis

The first burn was of insufficient duration to allow a meaningful analysis to be performed. In addition, the data from the gaging system during the first burn appeared questionable. Attempts were made to simulate the entire steady-state portion of the second burn. The results (correlation between model predicated value and actual data) were considered to be of an unsatisfactory nature. Special problems were encountered in attempting to model crossover. There was no consistency in the flight data between the storage and sump tank gaging system and system pressures as to when crossover occurred. In addition, the storage tank gaging system data again were questionable. (The gaging system is discussed in greater detail in a later section.) These problems precluded the possibility of accurately determining the performance prior to crossover. A satisfactory simulation was obtained during the steady-state period after storage tank depletion (crossover) to cut-off at the second burn. Attempts to simulate the entire second burn satisfactorily are being continued.

The results of the simulation using the APAP are contained in table II (identified as analysis results) and are shown in figures 10 through 13. The analysis was performed of the 170-second portion (08:12:40 or 29 560 seconds range time) from 105 to 275 seconds (08:15:30 or 29 730 seconds range time) of the second SPS burn following crossover. The values indicated in table II are for 185 seconds after the second SPS ignition (08:14:00 or 29 640 seconds range time) and are representative of the values throughout the period of the burn analyzed. These results are the best estimates of the actual conditions in flight and are not corrected to standard inlet conditions.

Figures 14 through 17 show analysis program output plots which represent the residuals or differences between the actual flight data and program calculated values. Also presented in these figures are the actual flight data. These figures represent acceleration, chamber pressure, oxidizer sump tank gaging system, and fuel sump tank gaging system, respectively. A strong indication of the accuracy of the analysis program simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the AGC ΔV data transmitted via measurement CG0001V. Figure 14 shows the thrust acceleration during the portion of the burn analyzed, as derived from the AGC data, and shows the residual error between the AGC and program calculated values. The residual error time history is seen to have essentially a zero mean and little, if any, discernible trend. This indicates that the simulation is relatively valid, although other factors must also be considered in giving a critique of the simulation. As shown in figure 15, the analysis program calculated a chamber pressure approximately 0.7 psia higher than that measured initially. The differences narrowed until both were about equal after 170 seconds. This difference is well within the accuracy of the transducer. Modifications made in the transducer mounting to correct problems associated with the thermal environment that appeared in the AS-202 data were apparently satisfactory. The sump tank oxidizer gaging system data exhibited close agreement with the calculated oxidizer depletion rate (fig. 1). There was a difference between the two fuel depletion values of about 40 pounds over the period analyzed (fig. 17).

The program simulation indicated that small biases existed in the oxidizer and fuel inlet pressure measurements of approximately 1.11 and 0.05 psia, respectively. The simulation verified the initial estimate of the spacecraft weight, requiring only a 31-pound adjustment to the Apollo Spacecraft Program Office (ASPO) supplied value. A sizable adjustment of -151 pounds oxidizer and -114 pounds fuel was made in the initial input of propellant weights for the analysis at 105 seconds into the second burn. Possible causes of this adjustment could be uncertainties in the propellant loading, uncertainties in the gaging system, uncertainties in propellant temperature (and thus propellant density), and possible errors in extrapolating from the initial loaded weights to the actual burn time analyzed.

As previously discussed, the lack of T/M propellant temperature measurements precludes an accurate determination of propellant densities, which decreases confidence in the simulation since propellant densities strongly affect the simulation. The lack of point sensor data (auxiliary propellant utilization and gaging subsystem (PUGS)) also compromises the simulation somewhat by magnifying one of the most difficult analysis problems, that of determining propellant flow rates.

A listing of the input deck to the APAP is provided in the appendix.

## Steady-State Performance Evaluation and Comparisons

The engine acceptance tests are conducted in order to determine the performance of the engine segregated from the feed system. This enables engines to be evaluated on their own merit, and provides a common basis on which to compare engines. It was determined from the analysis of the Apollo 4 flight that the SPS engine performance, corrected to standard inlet conditions, yielded a thrust of 21 503 pounds, a specific impulse of 312.06 seconds, and a mixture ratio of 2.007. The uncertainty ($3\sigma$) of these results is estimated to be 0.50 percent for thrust and specific impulse and 1.8 percent for mixture ratio. These values of thrust, specific impulse, and mixture ratio are 0.01 percent higher, 0.27 percent lower, and 0.37 percent higher, respectively, than reported in the Acceptance Test Log for Test Number 3.5-07-DPA-029 on engine serial number 0000031. These differences are within the expected ranges. The standard inlet conditions performance values reported herein were calculated for the following conditions:

1. Oxidizer interface pressure, 164 psia

2. Fuel interface pressure, 170 psia

3. Oxidizer interface temperature, 70° F

4. Fuel interface temperature, 70° F

5. Oxidizer density, 90.15 $lbm/ft^3$

6. Fuel density, 56.31 $lbm/ft^3$

7. Thrust acceleration, 1.0

8. Throat area (initial value), 121.66 in$^2$

The analysis program-calculated actual flight values are also compared in table II to the integrated feed/engine systems predicted flight values as reported in Internal Note MSC-EP-R-67-33, AS-501 Service Propulsion System Preflight Report, dated October 11, 1968. The analysis results of thrust, specific impulse, and engine mixture ratio were within 0.38, 0.07, and 1.00 percent, respectively, of the predictions. These differences are considered acceptable and well within the expected tolerances.

A similar analysis of the SPS performance for AS-202 was performed by TRW. The analysis program used was considered the predecessor of the present analysis program with basically the same techniques involved. A comparison of the specific impulses for the two flights are as follows:

| Spacecraft | Actual flight values, sec | Corrected to standard inlet conditions, sec | North American-Rockwell reported standard inlet condition values from acceptance test, sec |
|---|---|---|---|
| AS-202 | 310.4 | 311.54 | 312.8 |
| AS-501 | 311.76 | 312.06 | 312.9 |

The operational trajectory was generated using a constant steady-state thrust of 21 500 pounds, a specific impulse of 312.8 seconds, a mixture ratio of 2.0. These values are within 0.40 percent, 0.35 percent, and 0.50 percent, respectively, of the program calculated values during the portion of the long burn analyzed. Larger differences, however, especially in thrust, would exist prior to crossover since the operational trajectory model did not include the increase in thrust (approximately 600 pounds) that occurs at crossover.

## GAGING SUBSYSTEM ANALYSIS

The propellant utilization and gaging subsystem (PUGS) was operated in the primary mode during the Apollo 4 mission. A bias in the PUGS for the sump tanks exists because of a difference in liquid levels in the propellant sump tanks and inside the gaging system stillwell. The stillwell is, in essence, a manometer and balances the pressure at the bottom of the stillwell with a fluid head. Under nonflow conditions, this fluid head equals the level of propellant in the tank. However, when the propellant is flowing, the fluid head in the stillwell is reduced by the dynamic head of the propellant flowing by the bottom of the stillwell through the zero-g retention reservoir.

The storage tank gaging system reading at the beginning of the first burn was 200 and 275 pounds higher than the reported KSC pad values for oxidizer and fuel, respectively. The first burn was a no-ullage start. The storage tank gaging system was very erratic for both fuel and oxidizer during the first burn and could have been affected by propellant slosh in the storage tanks. The storage tank gaging system readings were the same for the first burn shutdown and second burn ignition. During the second burn (figs. 8 and 9), the storage tank gaging system for oxidizer exhibited "sawtooth" shifts and at depletion had a +100-pound bias. After depletion, a definite drift was observable.

The sump tank values agree with the reported KSC pad values when the dynamic bias is accounted for on the first burn. Between the first and second burns, shifts of +150 and +35 pounds were noted in the sump tank readings for oxidizer and fuel, respectively. After storage tank depletion (crossover), the sump tank gaging probes seem to indicate a lag in response. The gaging output was constant until approximately 4 seconds after crossover as indicated by engine inlet pressure increase. After the sump probes would start to respond, an unusually high propellant flow rate was indicated for about 16 seconds for oxidizer and 6 seconds for fuel. The values then stabilized and decreased linearly for the remainder of the burn. The observed characteristic is apparently caused by the fact that initially the levels in the sump tank are above the cylindrical section and are in the spherical part of the tank. Because of the dynamic flow bias, the probe senses a lower level which, based on the shaping of the probe, is associated with a larger cylindrical tank diameter. Since the probe is really sensing a height change, the apparent flow rate is high until the propellant levels reach the cylindrical section of the tanks.

During sump tank depletion, after stabilization, the gaging system results appeared normal. The oxidizer flow rate derived from the gaging system, after accounting for the bias change with acceleration, agreed with the performance calculated flow rate almost exactly and the fuel flow rate within 1.0 percent (figs. 16 and 17). The fuel gaging system values were lower than the calculated performance values.

The AS-202 analysis was performed using the auxiliary gaging system (point sensors). Study cases were made, however, using the primary gaging system in which plots comparable to figures 16 and 17 were generated. There was a striking similarity between the characteristics of the primary gaging system residuals for AS-202 and AS-501. From figures 16 and

17, about a 20-pound shift in the oxidizer gaging system at 29 645 seconds range time is evident. Similar shifts occurred in the AS-202 gaging system data at the same level on the probes. The slopes of the oxidizer and fuel residuals were also similar. This would indicate that the shift is not due to increased propellant usage but a hardware associated shift.

Tests conducted at White Sands Test Facility (WSTF) on both Block I and Block II test vehicles indicate that during propellant crossover (two tank loading), more propellant was being transferred to the sump tank from the storage tank than to the engine, with a resulting increase in propellant level in the sump tank. No adequate explanation for this phenomenon has been found. The sump tank gaging system before crossover during the Apollo 4 mission did not show a level rise, indicating that it is a problem peculiar to WSTF testing. There was a small rise in sump tank gaging read-out before crossover; however, it was due to the previously mentioned gaging bias in the sump tank. The bias is inversely proportional to acceleration. The acceleration increase during the burn reduces the bias which raises the level in the gaging system stillwell, indicating a higher reading. The actual propellant level, however, does not change.

The results of the PUGS analysis on AS-501 indicates that the operation was fairly normal with the exception of exhibited tendencies to drift or shift during "off" periods. Because of this, the most reliable information to be gained is propellant depletion during long burns rather than absolute propellant magnitudes.

## PRESSURIZATION SYSTEMS

Both SPS pressurization systems operated nominally throughout the mission. Helium bottle pressure and temperature data showed a constant, nominal helium consumption during both SPS burns and no indication of helium leakage during the intermediate coast period. Gaseous nitrogen bottle pressures and the propellant ball valve traces indicated that both gaseous nitrogen valve banks operated nominally during the two SPS burns and no pressure loss occurred during the coast period.

## ENGINE TRANSIENT ANALYSIS

An analysis of the start and shutdown transients was performed to determine the transient impulse and time-variant performance characteristics during the Apollo 4 mission and to ascertain the effectiveness of the no-ullage start. The results of this analysis, which encompassed the transient regimes for both SPS burns, are summarized in table III.

Engine acceptance test data, specification requirements, and previous spacecraft flight data were employed to provide better insight into the meaningfulness of the Apollo 4 flight test results and the applicability thereof to subsequent flight development missions and the lunar landing mission. Start and shutdown transient plots of chamber pressure are shown in figures 18 through 21.

All transient specification criteria appeared satisfied, except for chamber pressure overshoot during start and impulse repeatability for shutdown. The chamber pressure overshoot phenomenon, as will be discussed later in this section, appears to correlate with the rapid response rates which are characteristic of the Apollo 4 SPS bipropellant valves, rather than with the conditions imposed by a no RCS ullage settling start. The 995 lb-sec shutdown impulse repeatability determined for Apollo 4 appears at least partially explained by the uncertainty associated with determining the time of the manually directed shutdown command signal. Ascribing an uncertainty of 0.10 second to the receipt of signal by the bipropellant valve on the second burn would assure repeatability to within the ±300 lb-sec specification.

The techniques utilized in evaluating the SPS transient performance and behavior characteristics during the Apollo 4 mission are detailed in the ensuing text.

Due to a manual cut-off signal for the second burn, velocity gain data from the guidance system could be calculated only for the first burn cut-off. It was calculated to be 7.07 ft/sec referenced to a cut-off time of 12 502.56 (03:28:22.56) seconds. The estimated average vehicle weight at this time was 50 615 lbm. Impulse is defined as the thrust-time integral as follows

$$I = \int_{t_{c/o}}^{t_{F=0}} F\,dt \tag{1}$$

Inserting $F = ma$ and assuming the mass is approximately constant during cut-off, the following is obtained

$$I = m\int_{t_{c/o}}^{t_{F=0}} a\,dt = \frac{-W}{g_c}\left(V_{t_{c/o}} - V_{t_{F=0}}\right) \tag{2}$$

where $I$ = cut-off impulse, lbf-sec

$F$ = thrust, lbf

$t$ = time, sec

$m$ = total vehicle mass, lbm

$W$ = total vehicle weight, lbm

$a$ = thrust acceleration, ft/sec$^2$

$g_c$ = conversion factor, lbm ft/lbf-sec$^2$

$V$ = thrust velocity, ft/sec

From equation (2) the cut-off impulse for the first burn can be calculated as follows

$$I = \frac{50\ 615}{32.174}(7.07) = 11\ 120 \text{ lbf-sec}$$

Since velocity gain is not measured during engine starts, and since the velocity gain was not available for the second burn cut-off, the chamber pressure data were used to approximate the related impulses. The relation used is as follows

$$I = \int_{t_{c/o}}^{t_{F=0}} F dt = \int_{t_{c/o}}^{t_{F=0}} C_f P_c A_t dt = C_f A_t \int_{t_{c/o}}^{t_{F=0}} P_c dt \qquad (3)$$

where $C_f$ and $A_t$ are assumed constant during the transients.

$C_f$ = thrust coefficient (dimensionless)

$A_t$ = throat area, in$^2$

$P_c$ = chamber pressure, lbf/in$^2$

In calculating the transient impulses, the value of $C_f$, often used when transient thrust data are not available, is the steady-state $C_f$ value. The actual value of $C_f$ is greatly influenced by the mixture ratio and the chamber pressure, both of which are rapidly changing during the start and cut-off periods. Using the steady-state value of $C_f$ is thus admittedly poor. To improve the estimate of the transient impulse, the value of $C_f$ to be used for all other transients was determined by applying equation (3) to the cut-off impulse determined by the velocity gain during the first burn

$$C_f = \frac{I}{A_t \int_{t_{c/o}}^{t_{F=0}} P_c dt} = \frac{11\ 120}{121.57 \times 49.32} = 1.855$$

Applying equation (3), the following results were obtained.

1. First burn start impulse from FS-1 to 90 percent steady-state thrust

$$I = C_f A_t \int P_c dt$$

$$I = (1.855)(121.66)(0.8421)$$

$$= 190.0 \text{ lbf-sec}$$

The measured time interval was

$$\Delta t = 0.41 \text{ sec}$$

2. First burn cut-off impulse from FS-2 to 10 percent steady-state thrust

$$I = (1.855)(121.57)(47.43)$$

$$= 10\ 700 \text{ lbf-sec}$$

The measured time interval was

$$\Delta t = 0.80 \text{ sec}$$

3. Second burn start impulse from FS-1 to 90 percent steady-state thrust

$$I = (1.855)(121.57)(1.5748)$$

$$= 355.1 \text{ lbf-sec}$$

The measured time interval was

$$\Delta t = 0.35 \text{ sec}$$

4. Second burn cut-off impulse from FS-2 to 10 percent steady-state thrust

$$I = (1.855)(120.43)(56.7848)$$

$$= 12\ 680 \text{ lbf-sec}$$

The measured time interval was

$$\Delta t = 0.89 \text{ sec}$$

The average start impulse is 272.6 lbf-sec with an average time of 0.38 second. The average cut-off impulse is 11 690 lbf-sec with an average time of 0.85 second.

In generating operational trajectories the FS-1 to 90 percent start impulse is normally used along with the steady-state values and shutdown impulse. The impulse produced by the chamber pressure overshoot was also calculated. For the first burn the overshoot impulse was 1861 lbf-sec and 1040 lbf-sec for the second burn. These values are quite significant when compared to the start impulse, and it may be desirable to account for the additional overshoot impulse when generating operational trajectories, particularly for a series of short burns.

If the shutdown impulse is taken from FS-2 to thrust = 0.0, the following values are obtained.

1. First burn impulse from FS-2 to thrust = 0.0

$$I = (1.855)(121.57)(49.32)$$

$$= 11\ 120 \text{ lbf-sec}$$

2. Second burn impulse from FS-2 to thrust = 0.0

$$I = (1.855)(120.43)(58.52)$$

$$= 13.070 \text{ lbf-sec}$$

The chamber pressure overshoot during the first burn start transient was observed to be 49.5 percent above the nominal steady-state level. Current specifications limit this overshoot to 20 percent; however, it has been exceeded on all SPS flights. Table IV delineates the chamber pressure overshoot and valve response times denoted on these first three spacecraft development missions. It should be noted that the flight chamber pressure measurement is sampled at 100 samples per second and has a nominal range of 0 to 150 psia. It is possible that the maximum chamber pressure reading is not being indicated. On figure 18, two data samples at about 150 psia are at their maximum range. The "X's" shown on the plot have no significance and are a function of the machine plotter. The chamber pressure transducer was mounted differently for Spacecraft 017 than for the previous mission, and this could affect the pressure indications. A ground test program will be conducted to determine whether the indicated overshoot is partially due to instrumentation inaccuracies or whether it is totally characteristic of the start.

Analyses of the Apollo 4 first SPS start regimes indicate that the 49.5-percent chamber pressure overshoot did not appear to be linked to the no-ullage start. Although the second start, which was preceded by an ullage maneuver, was characterized by a 15-percent lower overshoot, this reduction appears explained by the slower responding valves on the latter burn. As observed, the table IV data indicate a reasonably linear correlation between response time of the leading valve and chamber pressure overshoot for the first burn transients of three missions. Normally, the numbers one and four valve travel times are $0.6\ ^{+0.2}_{-0.05}$ second (dry) with the numbers two and three valve times being 0.325 ± 0.1 second (dry). Since valves one and two are in series and the combination in parallel with the numbers three and four valves which are in series, the numbers one and four valves are normally controlling in terms of chamber pressure rise rate. As shown in table IV the number four valve response was significantly faster. It appears that this rapid response, in turn, incited the excessive overshoot. No final conclusions can be made regarding overshoot until the ground test study is completed.

## DETAILED TEST OBJECTIVES

As an integral phase of data acquisition processes necessary prior to committal of the SPS to subsequent manned flights and/or the lunar landing mission, detailed test objectives were outlined for the Apollo 4 mission. These objectives provided a systematic opportunity for acquisition of data to certify the SPS for manned operations and contribute to a successful lunar mission. Objectives peculiar to the propulsion subsystem were P3.2 (SPS No-Ullage Start) and P3.3 (SPS Long Duration Burn).[1]

An examination has been made of the Apollo 4 postflight analysis results, discussed in the preceding sections, in view of the success criteria (SC) specified in these objectives. Discussion of these criteria is presented in the following sections.

### No-Ullage Start

This objective was intended to demonstrate that a SPS start can be satisfactorily performed in a zero-g environment with no RCS ullage settling maneuver when the sump tanks are full. Five criteria were specified in the mission requirements for determining the success or failure of this objective. These requirements and the postflight analysis of the mission are shown in table V. Postflight analyses indicated that of the five success criteria, four were successful. The failure of the system to meet SC number 5 (Maximum Chamber Pressure Overshoot) is not considered to be significant since this criterion has been exceeded on all SPS flight starts (Transient Analysis section discussion). The magnitude of the overshoot is considered to be a function of the valve response times rather than the no-ullage start.

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[1]NASA Report, "Apollo 4 and 6 Mission Requirements (501/017/LTA-10R and 502/020/LTA-2R), Unmanned Supercircular Reentry," dated September 27, 1967.

For SC number 3, the interface pressures are not measured. From measured engine inlet pressures, the interface pressures were calculated. The specified interface pressures were incorrect and, in addition, represent values for after crossover. After making the necessary adjustments, the criterion was satisfied.

Flight test data for the first burn showed no symptoms of helium ingestion, demonstrating the effectiveness of the propellant retention screens and the zero-g retention reservoirs to satisfactorily maintain propellants over the sump tank outlets (feedline inlets).

It should be noted that because of loading problems (Propellant Loading section), the ullage volume existing at the no RCS ullage settling start was less than planned, particularly for oxidizer, where the ullage volume was reduced by approximately 2 cubic feet from nominal. This means that the no RCS ullage settling start test was not conducted under the most severe conditions.

### Service Propulsion Subsystem Long-Duration Burn

This objective has as its purpose the determination of the effect of burn duration on SPS performance. Four success criteria were specified in the mission requirements for the evaluation of the success or failure of this objective. These data are compared to the results of the flight in table VI. Based on the analysis of these data, all success criteria were satisfied except SC number 3. The PUGS accuracy was not satisfied because of a tendency to drift for both oxidizer and fuel systems. In addition, the indicated fuel sump depletion rate was in error by 1 percent.

## CONCLUDING REMARKS

The performance of Spacecraft 017 service propulsion subsystem during the Apollo 4 mission was analyzed using the Apollo Propulsion Analysis Program. A satisfactory correlation of the flight data was achieved only for the portion of the burn after crossover (storage tank depletion). The resulting engine flight performance obtained from the analysis program for after crossover, more specifically at 08:14:00 range time, was determined to be as follows:

1. Thrust — 21 414 pounds

2. Specific impulse — 311.76 seconds

3. Mixture ratio — 2.014

These values, corrected to standard inlet conditions, resulted in the following values:

1. Thrust — 21 503 pounds

2. Specific impulse — 312.06 seconds

3. Mixture ratio — 2.007

These values are 0.01 percent higher, 0.27 percent lower, and 0.37 percent higher, respectively, than the acceptance test log results (also reported at standard inlet conditions). The uncertainty ($3\sigma$) of the above analysis results is estimated to be 0.50 percent for thrust and specific impulse and 1.8 percent for mixture ratio. The uncertainty of the acceptance test values is even greater thus the agreement is well within the expected tolerances.

The Apollo 4 service propulsion subsystem operation was nominal and the two detailed test objectives, no ullage start and long-duration burn, were accomplished to a satisfactory degree.

TABLE I.- FLIGHT DATA USED IN ANALYSIS PROGRAM

| Measurement number | Description | Nominal data range |
|---|---|---|
| SP0009P | Pressure, main valve engine oxidizer inlet | 0 to 300 psia |
| SP0010P | Pressure, main valve engine fuel inlet | 0 to 300 psia |
| SP0655Q | Quantity, oxidizer tank 1 primary | 0 to 16 000 lb |
| SP0656Q | Quantity, oxidizer tank 2 primary | 0 to 16 000 lb |
| SP0657Q | Quantity, fuel tank 1 primary -0 to 8000 lb | 0 to 8000 lb |
| SP0658Q | Quality, fuel tank 2 primary | 0 to 8000 lb |
| SP0661P | Pressure, engine chamber | 0 to 8000 lb |
| CG0001V | Computer digital data 40 bits | 0 to 150 psia |

TABLE II.- SERVICE PROPULSION SUBSYSTEM SYSTEM PERFORMANCE SUMMARY, APOLLO 4 MISSION

| Measurement and performance parameters | Value | | | | | |
|---|---|---|---|---|---|---|
| | SPS first burn | | | SPS second burn (a) | | |
| | Nominal | Measured | Preflight prediction (c) | Measured | Analysis results (b) | Preflight prediction (c) |
| Measurement description | | | | | | |
| SP0003 — oxidizer tank pressure, psia . . . . | Approx 181 | 175 | 181 | 176 | 183.2 | 181 |
| SP0006 — fuel tank pressure, psia . . . . . . | Approx 181 | 180 | 181 | 175 | 182.6 | 181 |
| SP0009 — oxidizer inlet pressure, psia . . . | Approx 154 | 153.5 | 152 | 158.8 | 157.9 | 156 |
| SP0010 — fuel inlet pressure, psia . . . . . | Approx 154 | 154.0 | 155 | 158.6 | 158.5 | 158 |
| SP0661 — engine chamber pressure, psia . . . | Approx 100 | 96.0 | 98.2 | 101.0 | 101.3 | 99.9 |
| Calculated performance parameters | | | | | | |
| Oxidizer flow rate, lb/sec . . . . . . . . . . . | Approx 45.8 | -- | 44.56 | -- | 45.89 | 45.57 |
| Fuel flow rate, lb/sec . . . . . . . . . . . . . | Approx 22.9 | -- | 22.67 | -- | 22.79 | 22.91 |
| Propellant mixture ratio . . . . . . . . . . . . | 2.00 ± 1 percent | -- | 1.96 | -- | 2.014 | 1.99 |
| Vacuum specific impulse, sec . . . . . . . . . . | 313 min. | -- | 311.7 | -- | 311.76 | 311.5 |
| Vacuum thrust, lb . . . . . . . . . . . . . . . | 21 500 ± 1 percent | -- | 20 958 | -- | 21 414 | 21 334 |

[a]After propellant crossover.

[b]Actual values from Apollo Propulsion Flight Analysis Computer Program at 08:14:00 range time.

[c]Internal note MSC-EP-R-67-33 — AS-501 SPS Preflight Report, 11 October 1967.

TABLE III.- SERVICE PROPULSION SUBSYSTEM ENGINE TRANSIENT ANALYSIS SUMMARY

| Parameter | AS-501 first burn | AS-501 second burn | AS-501 SPS engine 031 acceptance test | AS-202 first burn | AS-202 fourth burn | Specification values |
|---|---|---|---|---|---|---|
| Start transient total vacuum impulse from FS-1 to 90 percent steady-state thrust, lbF-sec . . . . . . . . . . . . | 190.0 | 355.1 | 227 | -- | -- | 100 to 400 |
| Time from FS-1 to 90 percent steady-state thrust, sec . . . | 0.41 | 0.35 | 0.351 | -- | -- | 0.350 to 0.550 |
| Engine run-to-run start repeatability, lbF-sec . . . . . . . | 273 ± 80 | 273 ± 80 | -- | -- | -- | ±100 |
| Shutdown transient total vacuum impulse from FS-2 to 10 percent steady-state thrust, lbF-sec . . . . . . . . | 10 700 | 12 680 | 9450 | -- | -- | 8000 to 13 000 |
| Time from FS-2 to 10 percent steady-state thrust, sec . . . | 0.80 | 0.89 | 0.751 | -- | -- | 0.650 to 0.900 |
| Engine run-to-run shutdown repeatability, lbF-sec . . . . . | 11 690 ± 995 | 11 690 ± 995 | -- | -- | -- | ±300 |
| Shutdown transient total vacuum impulse from FS-2 to 0 percent thrust, lbF-sec . . . . . | 11 120 | 13 070 | -- | 10 700 | 10 000 | -- |
| Time from FS-2 to 0 percent thrust, sec . . . . . . . . . . | 1.79 | 1.50 | -- | -- | -- | -- |

TABLE IV.- SERVICE PROPULSION SUBSYSTEM BIPROPELLANT VALVE RESPONSE AND CHAMBER PRESSURE OVERSHOOT CHARACTERISTICS

| Mission | Burn | Bipropellant valve response time, sec | | | | Chamber pressure Overshoot, percent |
|---|---|---|---|---|---|---|
| | | Valve no. 1 (SP0022) | Valve no. 2 (SP0023) | Valve no. 3 (SP0024) | Valve no. 4 (SP0025) | |
| AS-201 | First | 0.62 | 0.50 | 0.41 | 0.60 | 32.7 |
| | Second | (a) | (a) | (a) | (a) | (a) |
| AS-202 | First | .60 | .20 | .30 | .50 | 39.0 |
| | Second | .70 | .30 | .30 | .80 | 25.0 |
| | Third | .80 | .30 | .42 | .80 | 26.0 |
| | Fourth | (b) | (b) | (b) | (b) | (b) |
| Apollo 4 | First | .60 | .40 | .38 | .45 | 49.5 |
| | Second | .65 | .60 | .50 | .50 | 34.5 |

[a]Helium ingestion precluded a meaningful evaluation of this regime.

[b]Data acquired at ignition were invalid.

TABLE V.- SERVICE PROPULSION SUBSYSTEM NO-ULLAGE START OBJECTIVE

| Success criteria | Test value | Remarks |
| --- | --- | --- |
| 1. Start transient total impulse from onset of electrical command to 90 percent steady-rated thrust must be within Master End Item Specification SID 64-1237, paragraph 3.4.1.3.4.1.4.7.2. | I = 190 lbF-sec | Passed |
| 2. The engine must develop 90 percent steady-state thrust within 0.4 to 0.6 second after onset of the electrical command signal to the pilot valve. | $\Delta t$ = 0.41 | Passed |
| 3. During starting, the fuel and oxidizer pressures are within 6 psi of each other. During steady-state engine operation, the fuel is furnished to the propellant interface at 163 ± 4 psia and the oxidizer is furnished to the propellant interface at 160 ± 4 psia.[a] | $\Delta P$ = 2.6 psia<br>Interface pressures not available | Passed |
| 4. The steady-state thrust and mixture ratio, extrapolated to reflect nominal engine valve inlet propellant supply conditions, are to be within ±1 percent of 21 500 pounds and 2.00, respectively. | -- | Passed |
| 5. The transient starting chamber pressure is not greater than 120 percent of nominal chamber pressure. | $P_{Max}$ = 149 percent | Failed |

[a]Values specified are incorrect. True values are fuel 170 ± 4 psia and oxidizer 164 ± 4 psia.

TABLE VI.- SERVICE PROPULSION SUBSYSTEM LONG-DURATION BURN OBJECTIVE

| Success criteria | Test value | Remarks |
|---|---|---|
| 1. During steady-state engine operation, the fuel is furnished to the propellant interface at 163 ± 4 psia and the oxidizer is furnished to the propellant interface at 160 ± 4 psia.[a] | $P_{IO}$ = 163.9<br>$P_{IF}$ = 169.4 | Passed<br>Passed |
| 2. The steady-state thrust and mixture ratio, extrapolated to reflect nominal engine valve inlet propellant supply conditions, are to be within ±1 percent of 21 500 pounds and 2.00, respectively. | F = 21 503<br>MR = 2.007 | Passed<br>Passed |
| 3. Propellant utilization gaging system accuracy (after correction for PUGS bias) must be within 0.35 percent of full tankage capacity +0.35 percent of propellant remaining (applies separately to oxidizer and fuel). | See discussion | Failed |
| 4. Shutdown impulse is to be within 8000 to 13 000 lb-sec. | 12 690 | Passed |

[a]Values specified are incorrect. True values are fuel 170 ± 4 psia and oxidizer 164 ± 4 psia.

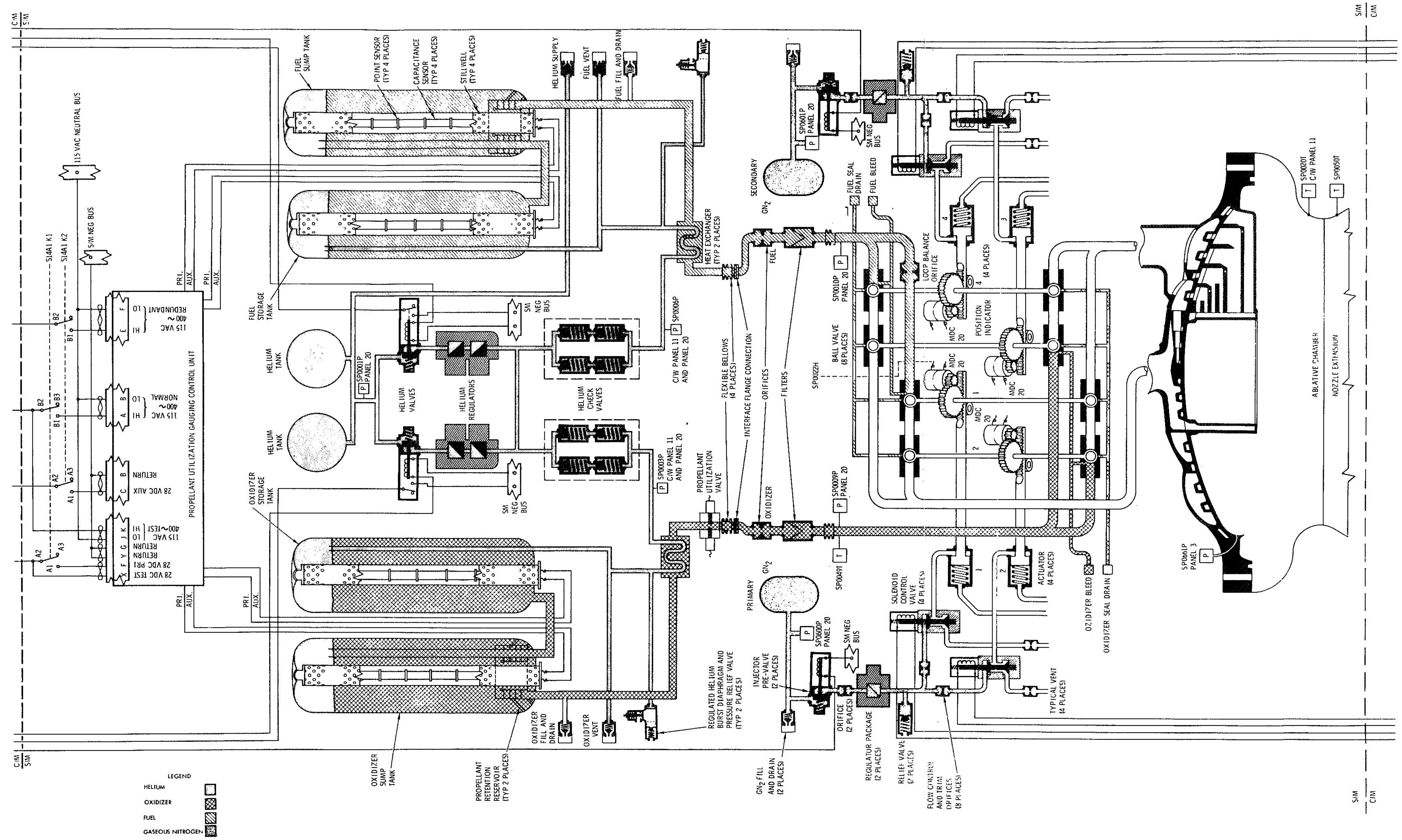


Figure 1.- Service propulsion subsystem configuration.

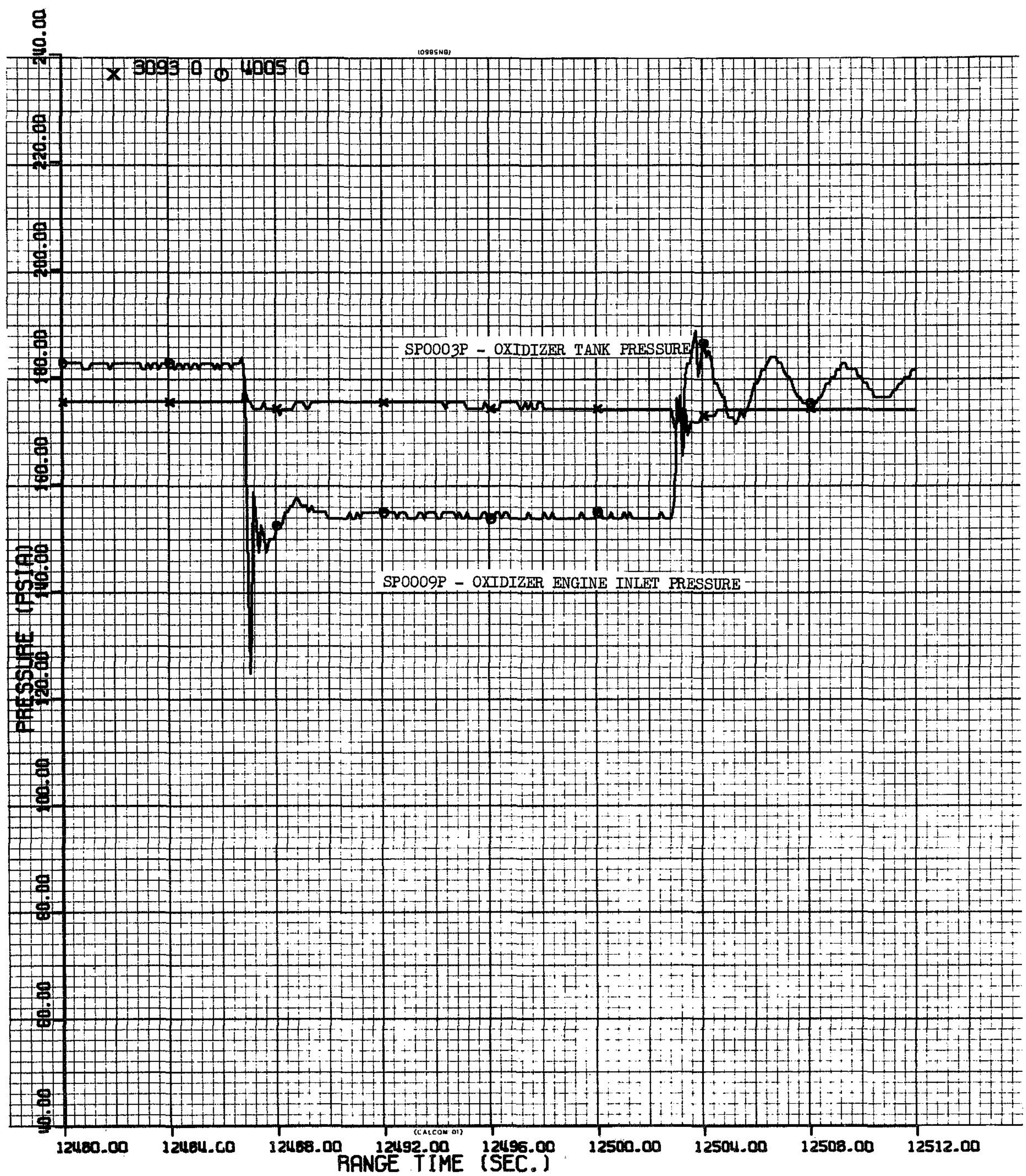



Figure 2. - Oxidizer system pressures — first burn.

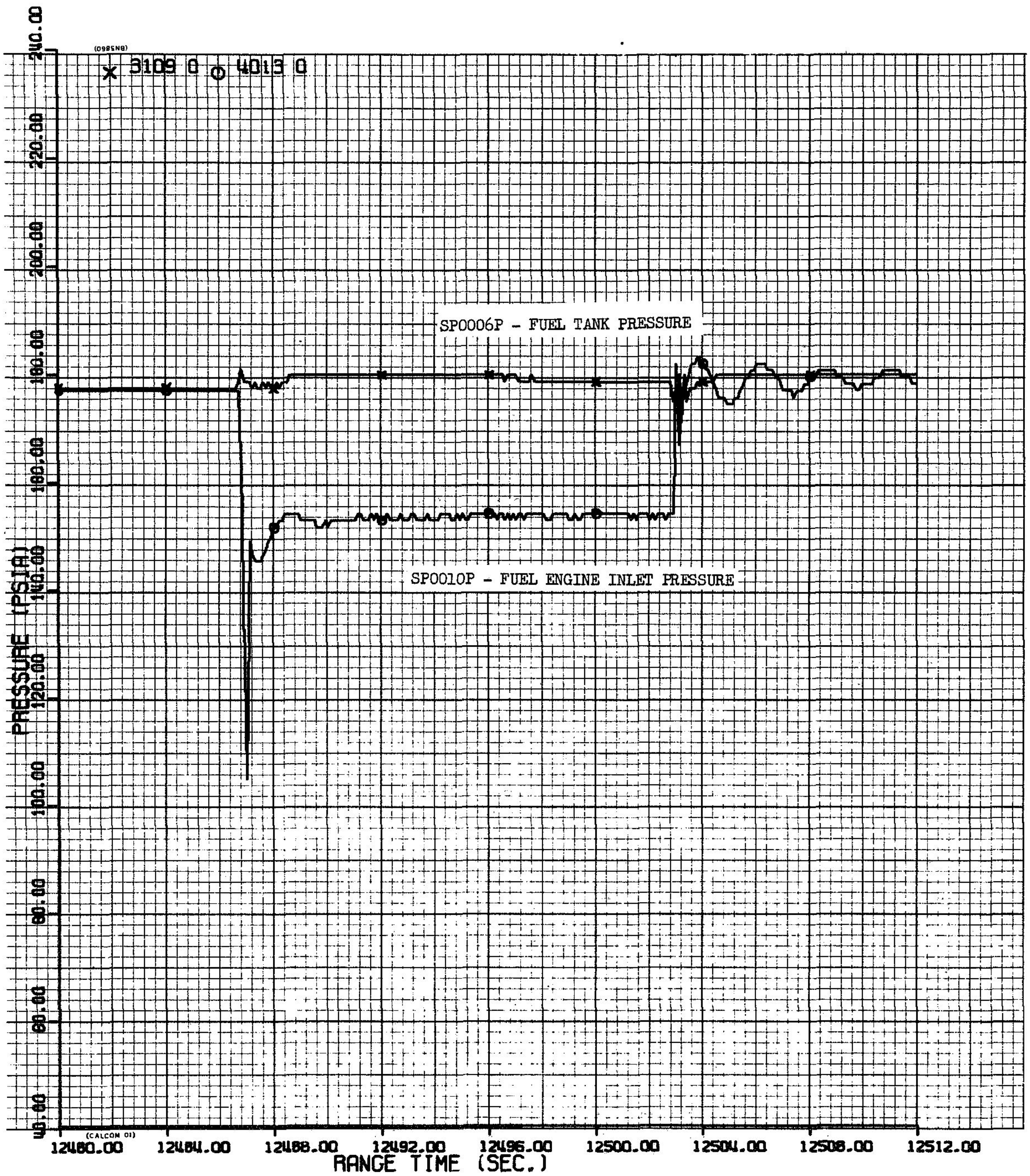



Figure 3. - Fuel system pressures — first burn.

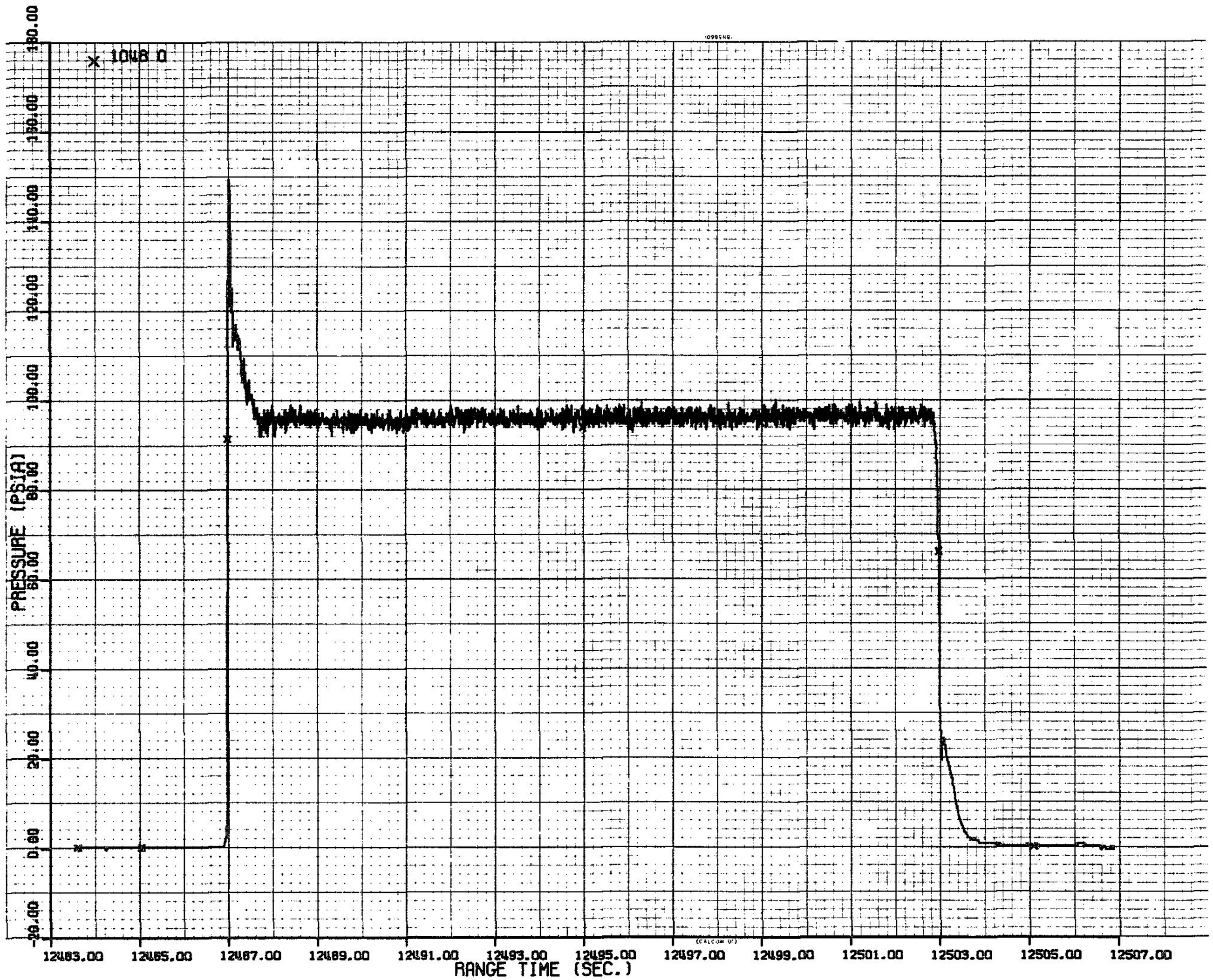
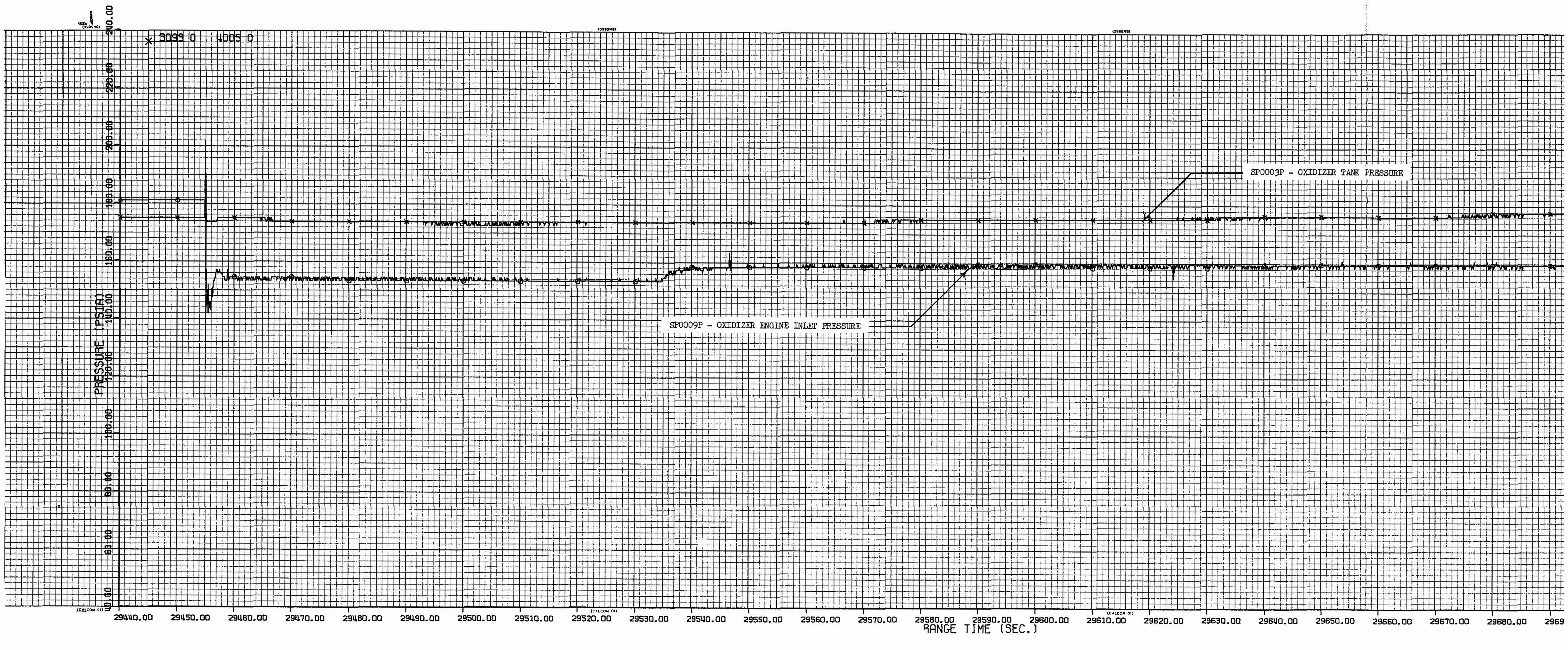



Figure 4. - Engine chamber pressure — first burn.



PRESSURE (PSIA)
RANGE TIME (SEC.)
SP0003P - OXIDIZER TANK PRESSURE
SP0009P - OXIDIZER ENGINE INLET PRESSURE
× 3095 0
4005 0


Figure 5.

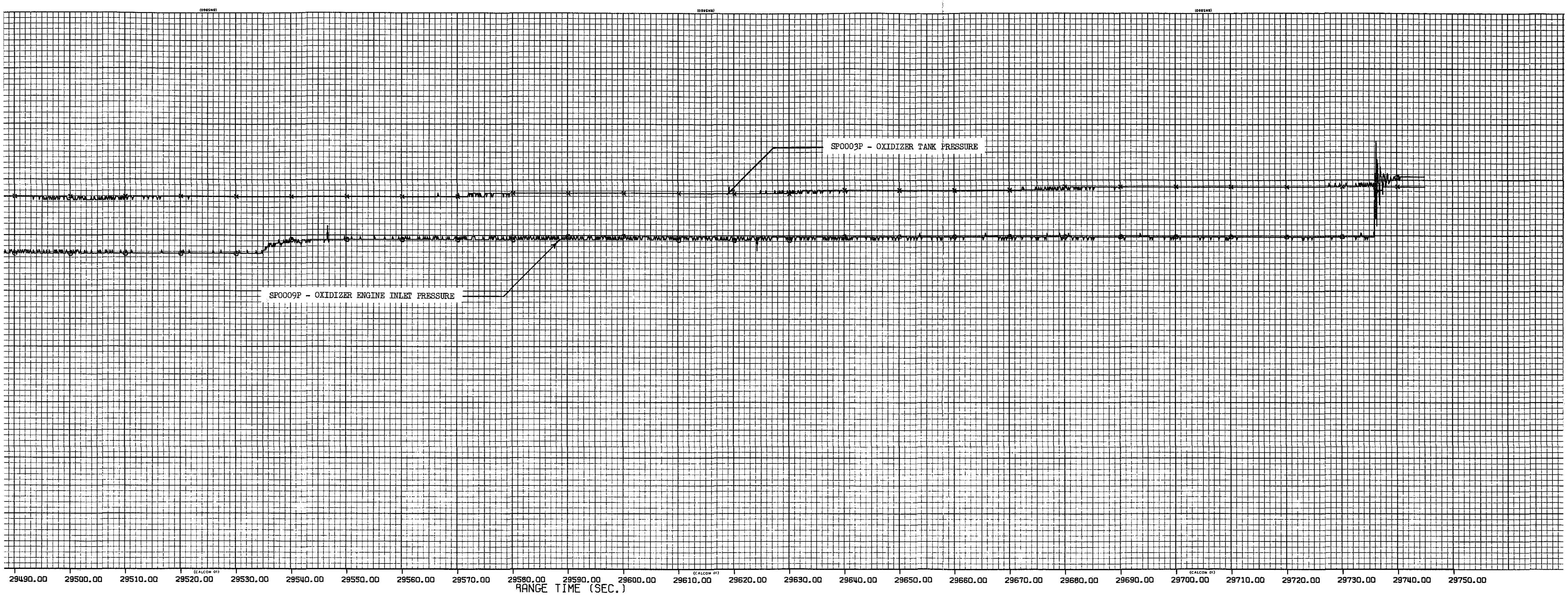
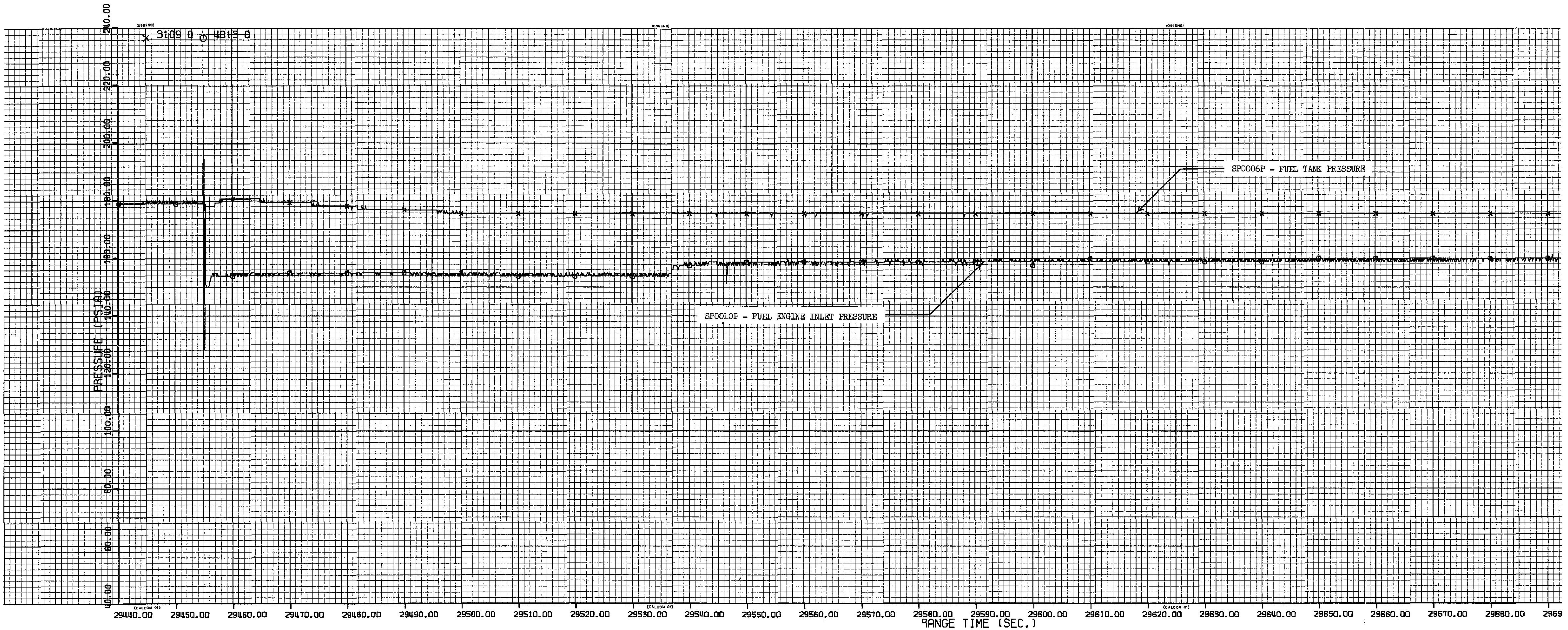



Figure 5.- Oxidizer system pressures — second burn.



× 3109 0 ○ 4013 0
PRESSURE (PSIA)
40.00
60.00
80.00
100.00
120.00
140.00
160.00
180.00
200.00
220.00
240.00
SP0006P - FUEL TANK PRESSURE
SP0010P - FUEL ENGINE INLET PRESSURE
29440.00
29450.00
29460.00
29470.00
29480.00
29490.00
29500.00
29510.00
29520.00
29530.00
29540.00
29550.00
29560.00
29570.00
29580.00
29590.00
29600.00
29610.00
29620.00
29630.00
29640.00
29650.00
29660.00
29670.00
29680.00
RANGE TIME (SEC.)


Figure

2



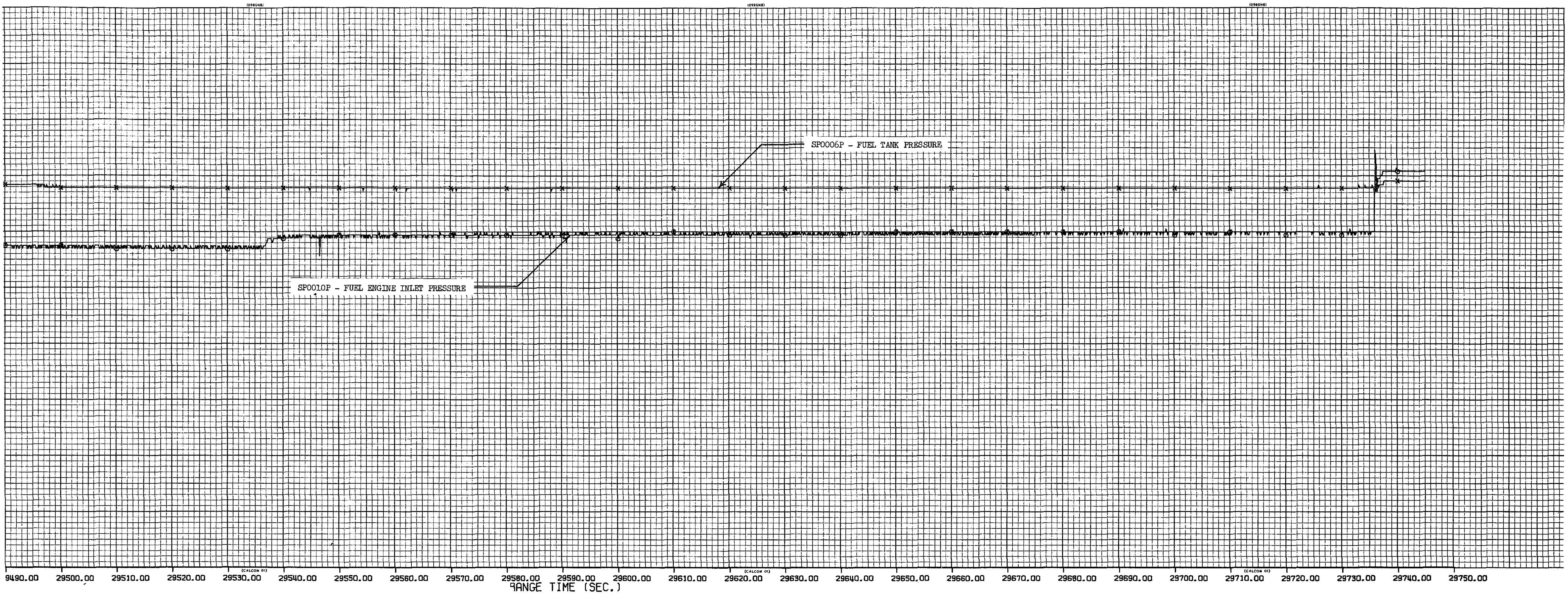
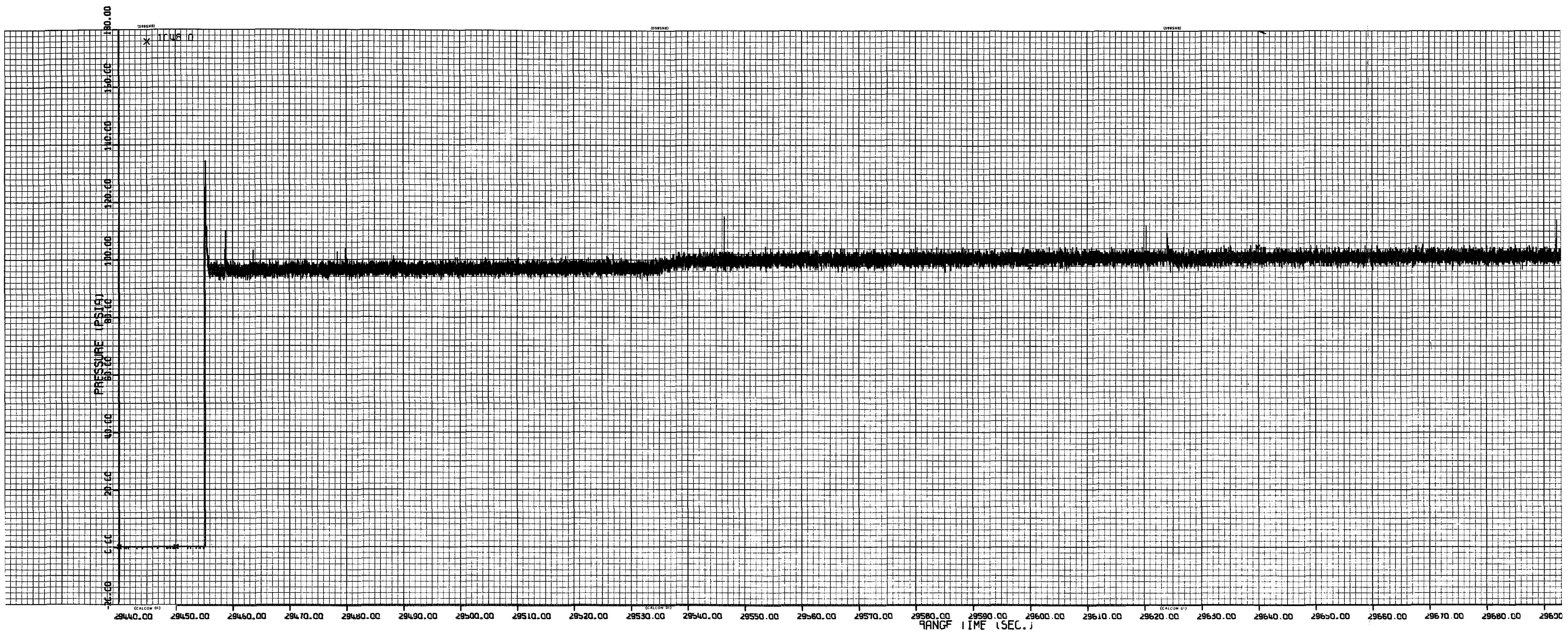



Figure 6.- Fuel system pressures — second burn.



1C48 0
PRESSURE (PSIA)
180.00
160.00
140.00
120.00
100.00
80.00
60.00
40.00
20.00
0.00
-20.00
29440.00
29450.00
29460.00
29470.00
29480.00
29490.00
29500.00
29510.00
29520.00
29530.00
29540.00
29550.00
29560.00
29570.00
29580.00
29590.00
29600.00
29610.00
29620.00
29630.00
29640.00
29650.00
29660.00
29670.00
29680.00
RANGE TIME (SEC.)


Figure 7

2



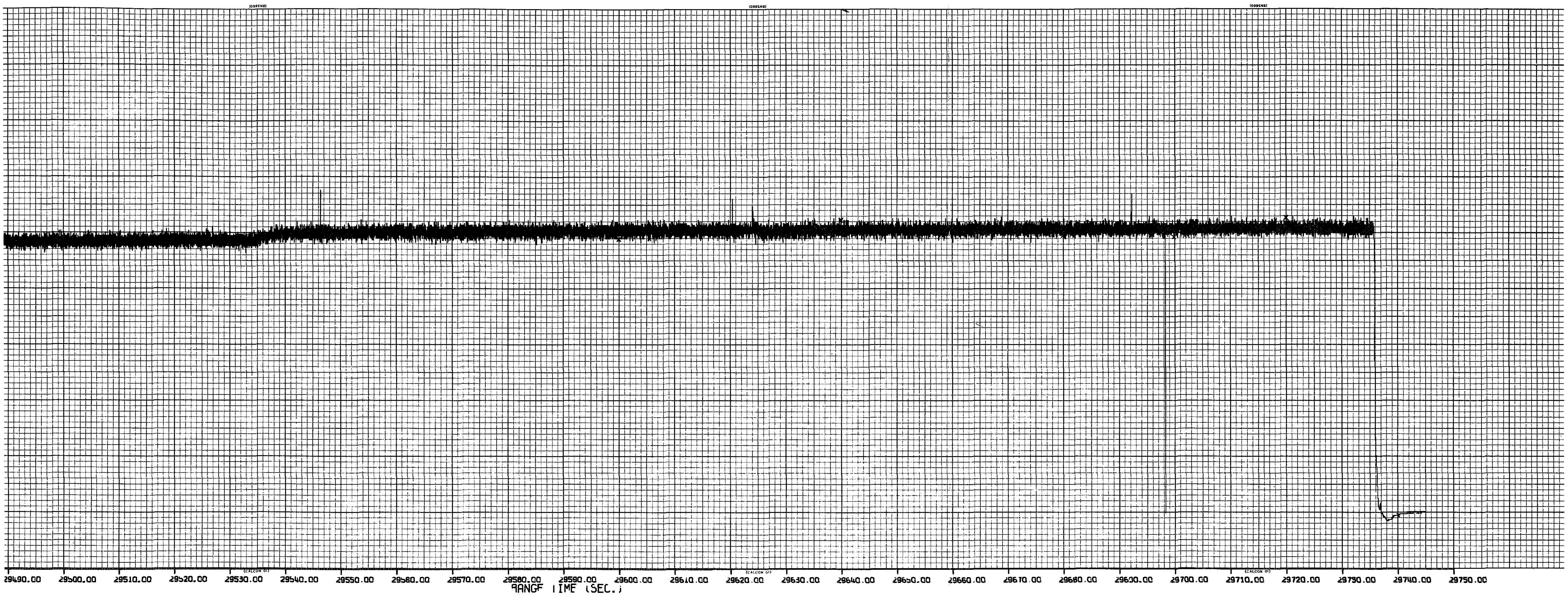
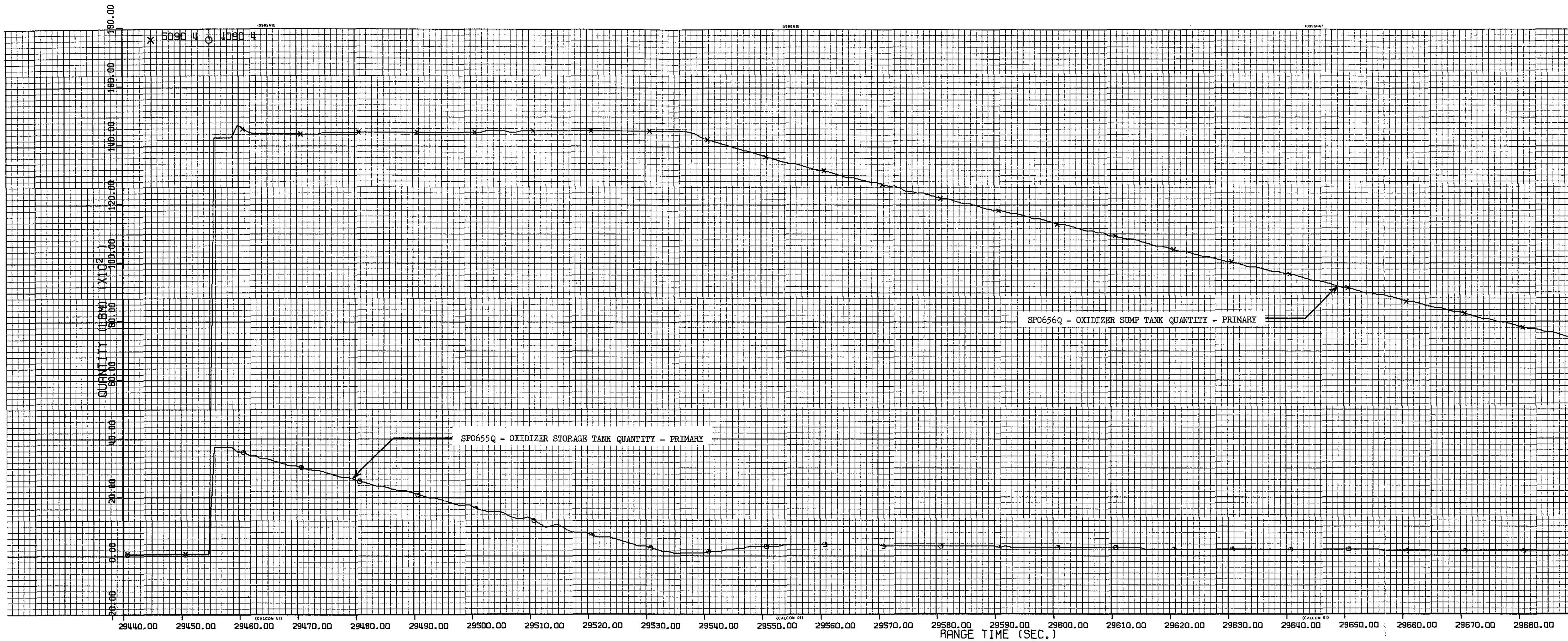



Figure 7.- Engine chamber pressure — second burn.



× 5090 4 ⊙ 4090 4
QUANTITY (LBM) (X10²)
180.00
160.00
140.00
120.00
100.00
80.00
60.00
40.00
20.00
0.00
-20.00
SP0656Q - OXIDIZER SUMP TANK QUANTITY - PRIMARY
SP0655Q - OXIDIZER STORAGE TANK QUANTITY - PRIMARY
29440.00
29450.00
29460.00
29470.00
29480.00
29490.00
29500.00
29510.00
29520.00
29530.00
29540.00
29550.00
29560.00
29570.00
29580.00
29590.00
29600.00
29610.00
29620.00
29630.00
29640.00
29650.00
29660.00
29670.00
29680.00
RANGE TIME (SEC.)


Figure 8.- 0

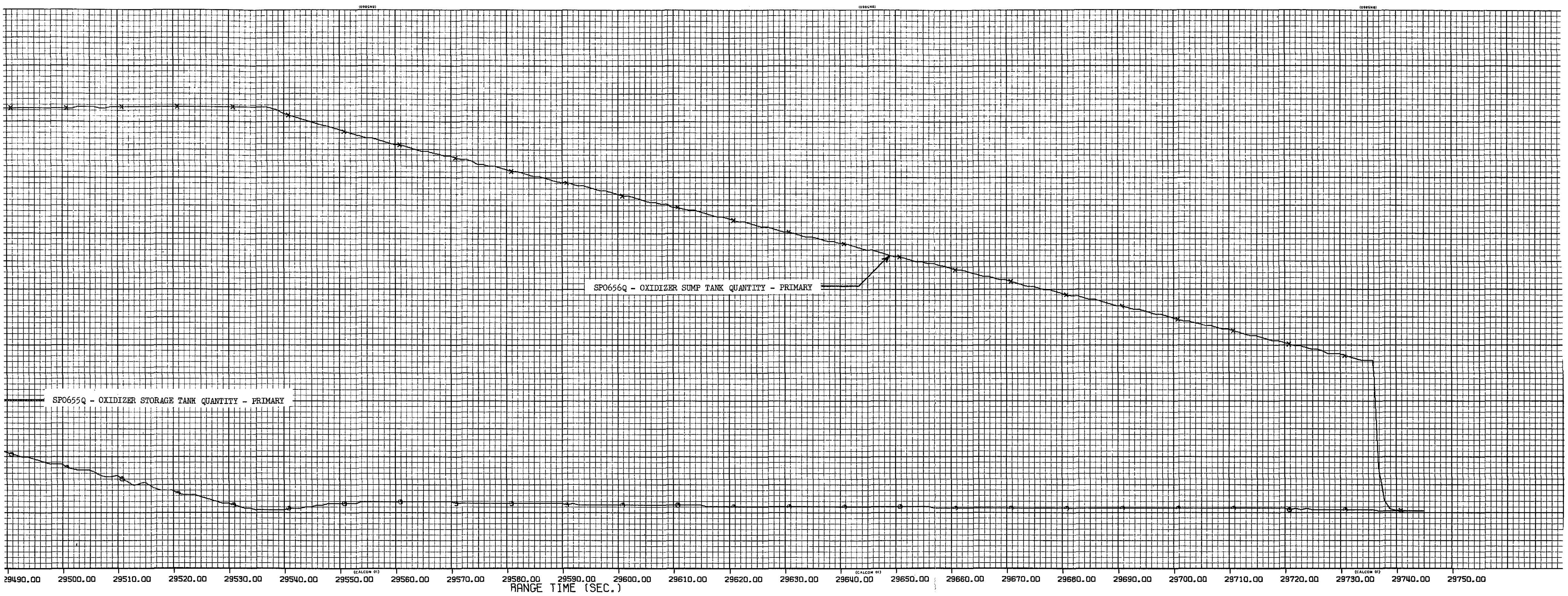
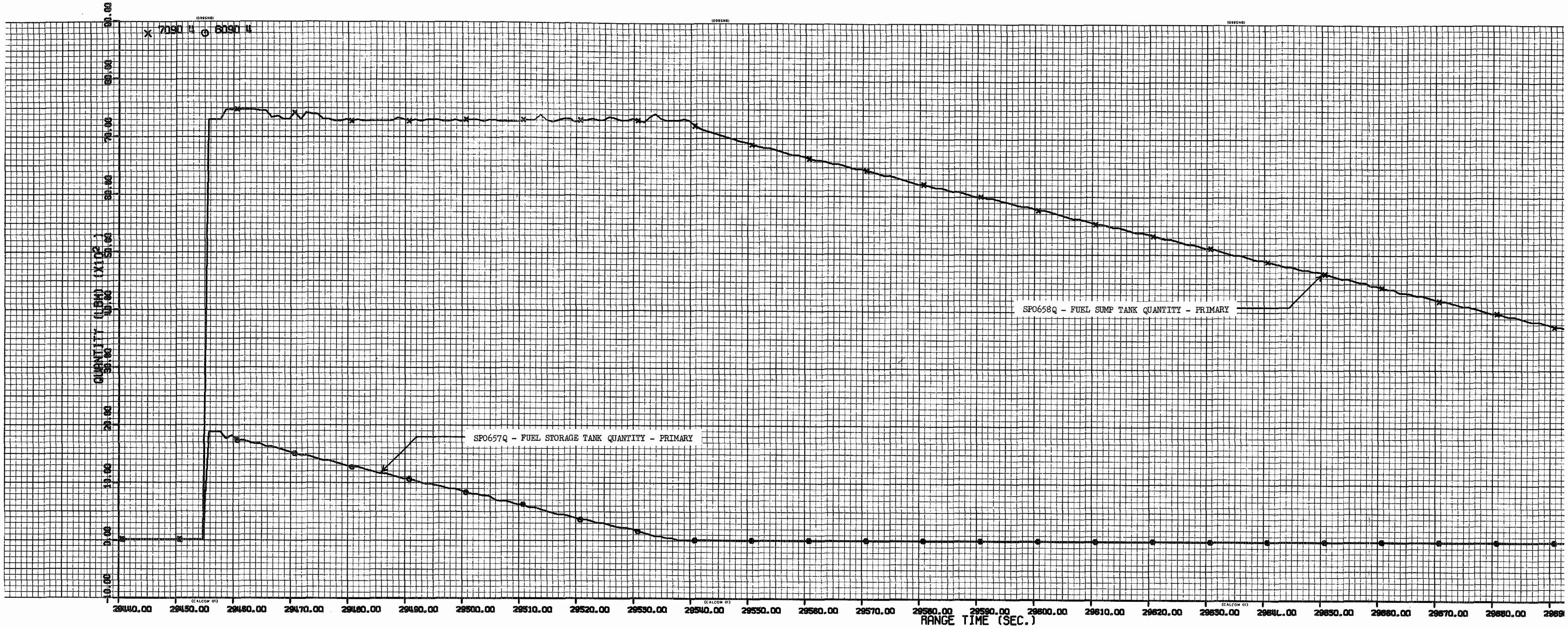



Figure 8.- Oxidizer primary gaging quantities — second burn.



× 7090 4 ⊙ 6090 4
QUANTITY (LBM) (X10^2)
SP0658Q - FUEL SUMP TANK QUANTITY - PRIMARY
SP0657Q - FUEL STORAGE TANK QUANTITY - PRIMARY
RANGE TIME (SEC.)


Figure 9.-

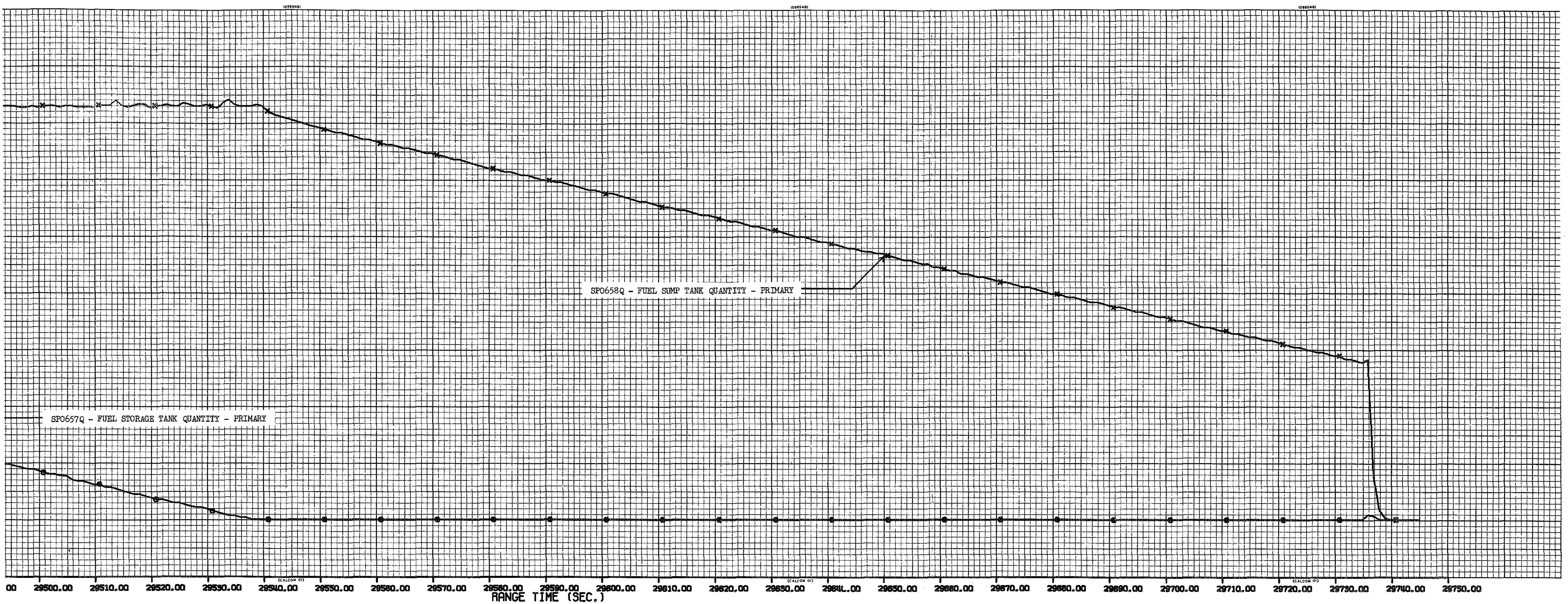



Figure 9.- Fuel primary gaging quantities — second burn.

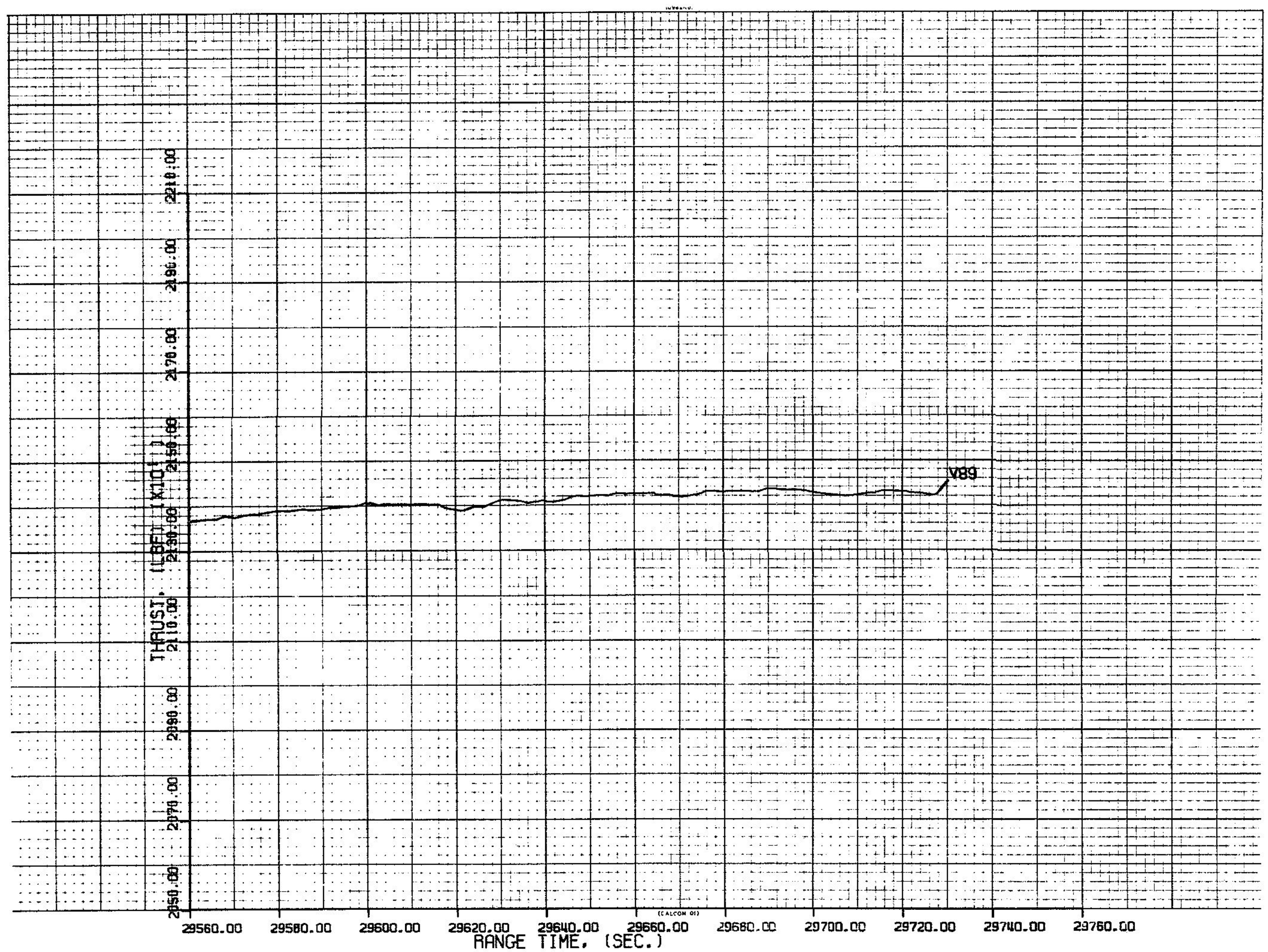



Figure 10. - Vacuum thrust — second burn (after crossover).

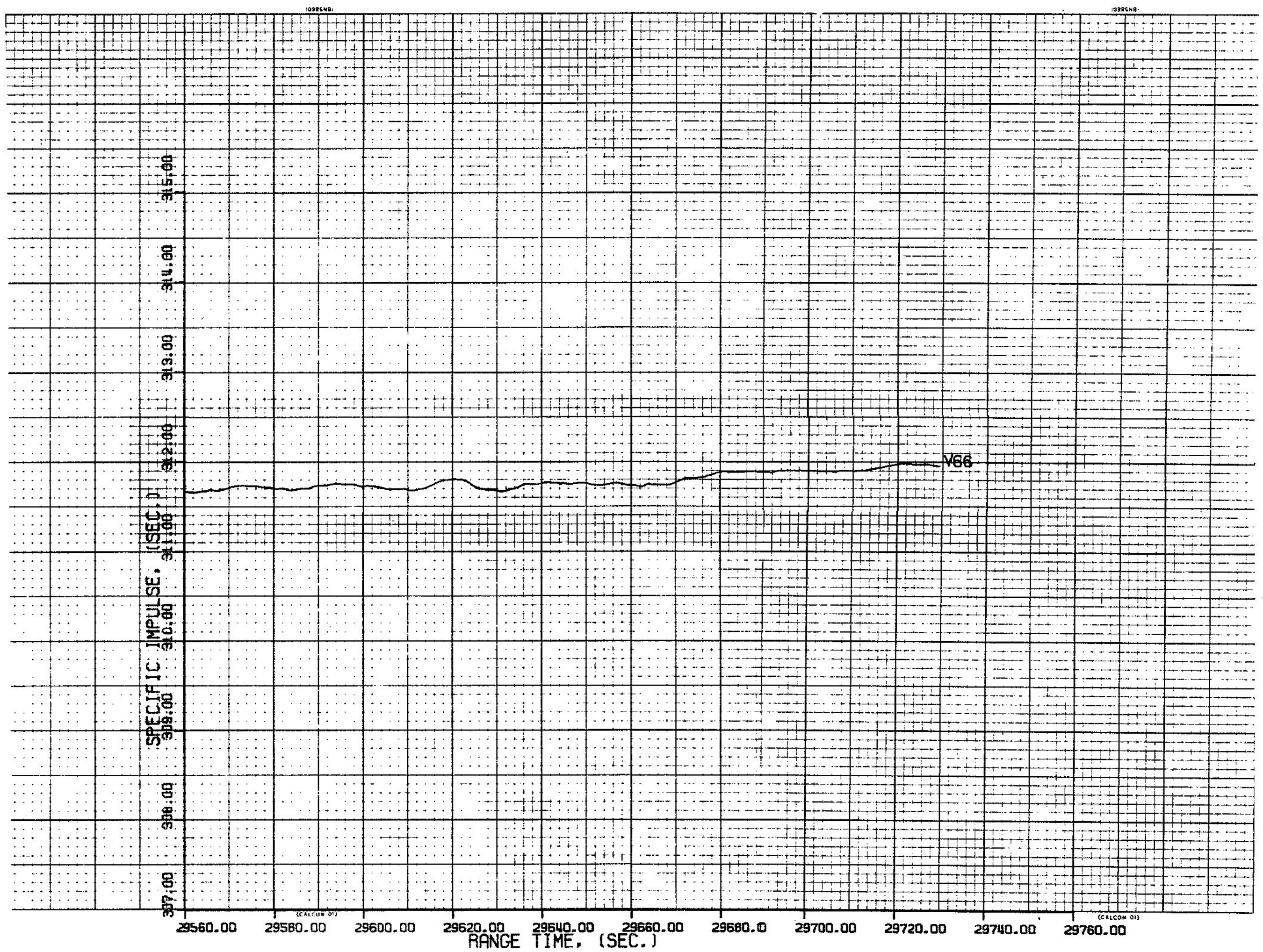



Figure 11. - Vacuum specific impulse — second burn (after crossover).

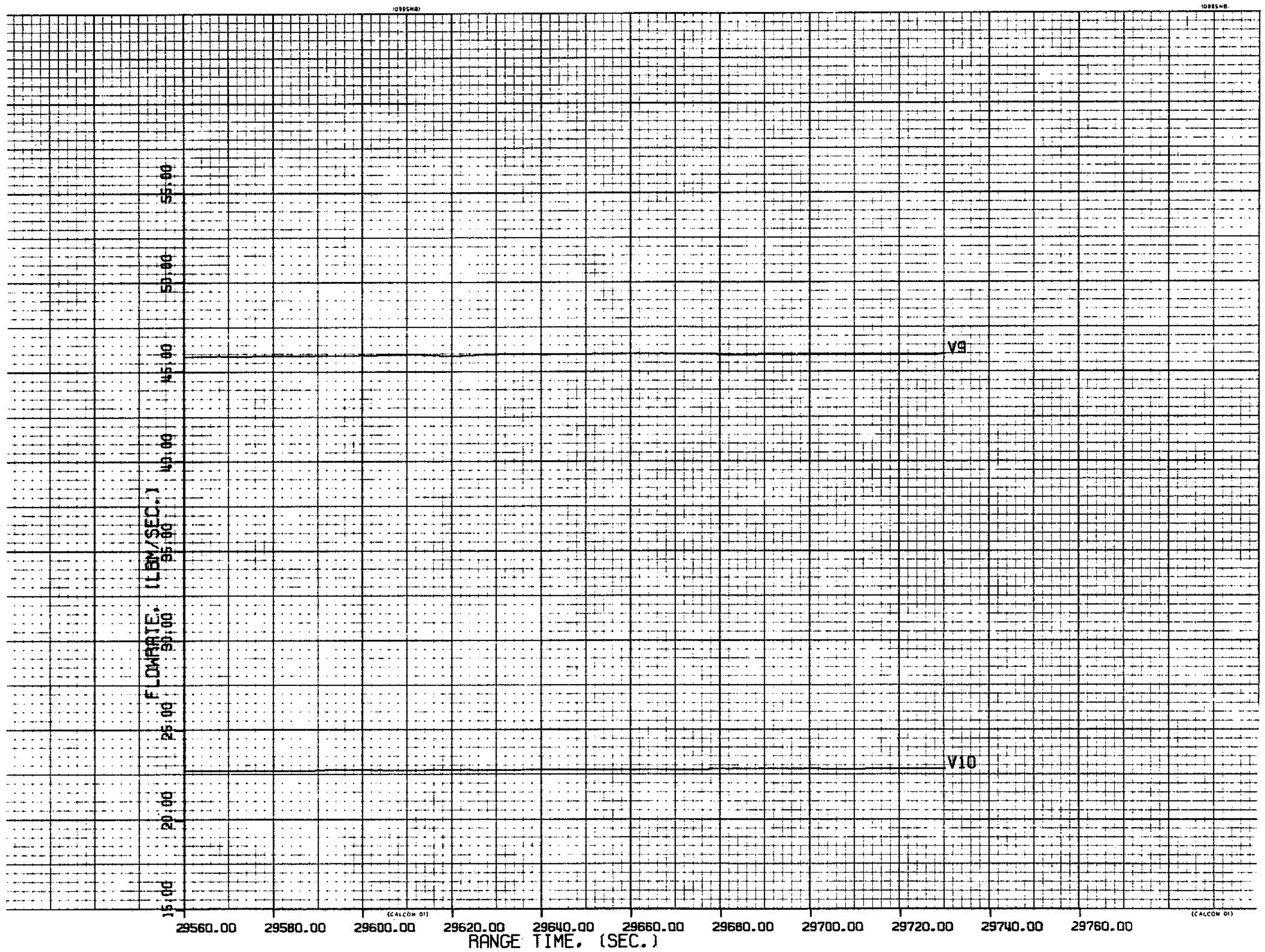



Figure 12. - Propellant flow rates — second burn (after crossover).

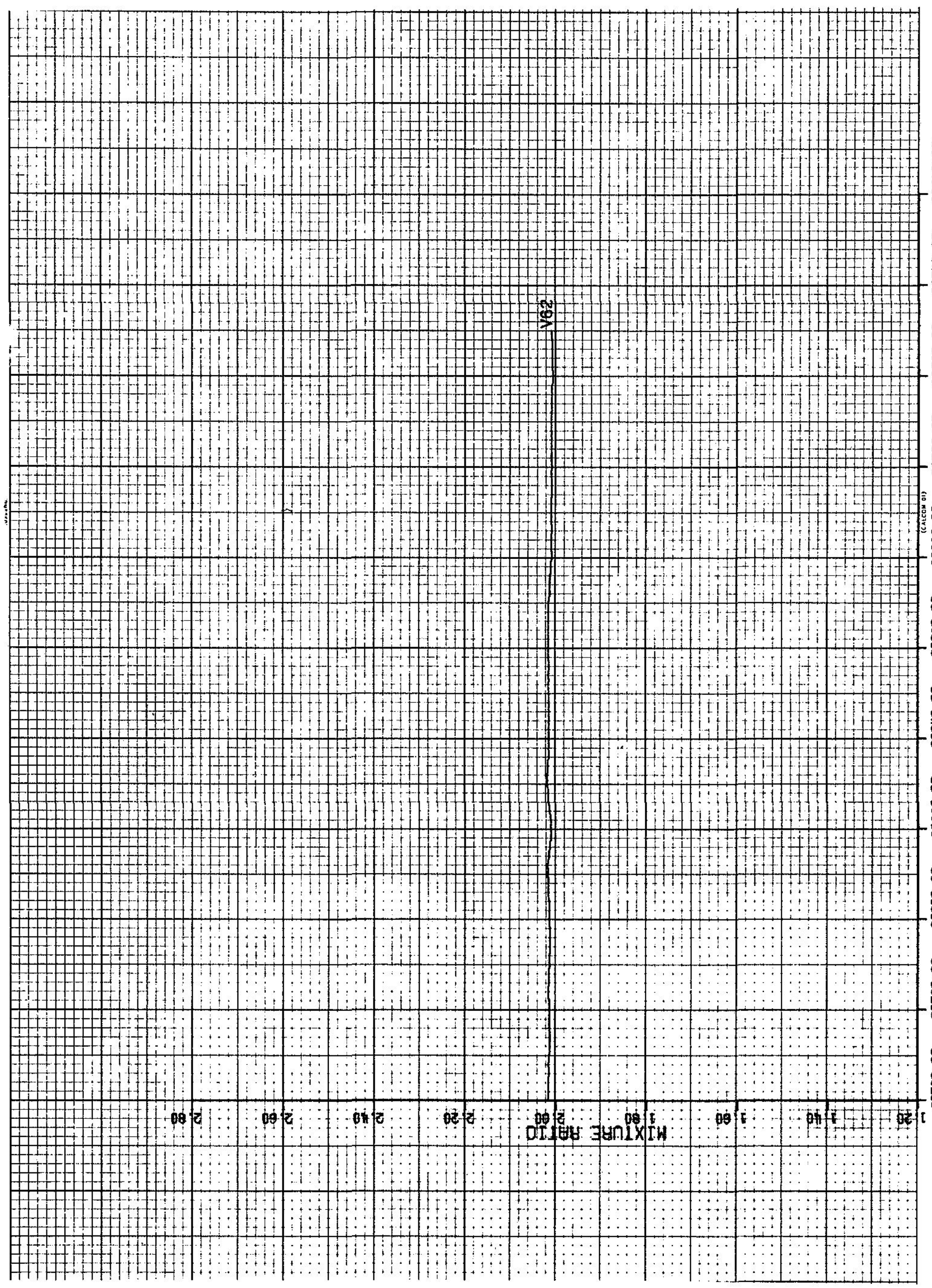



Figure 13. - Propellant mixture ratio — second burn (after crossover).

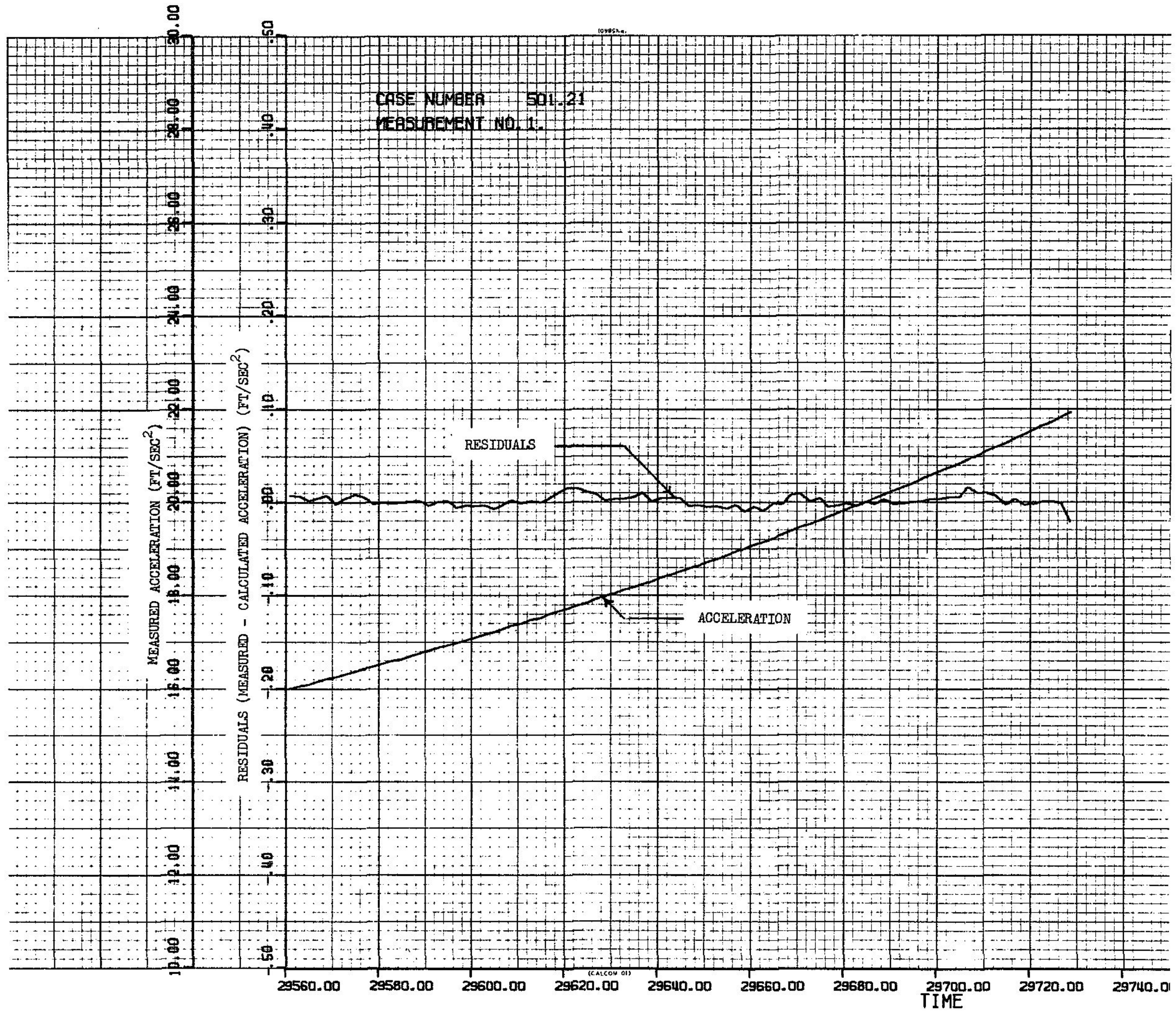



Figure 14. - Acceleration match — second burn (after crossover).

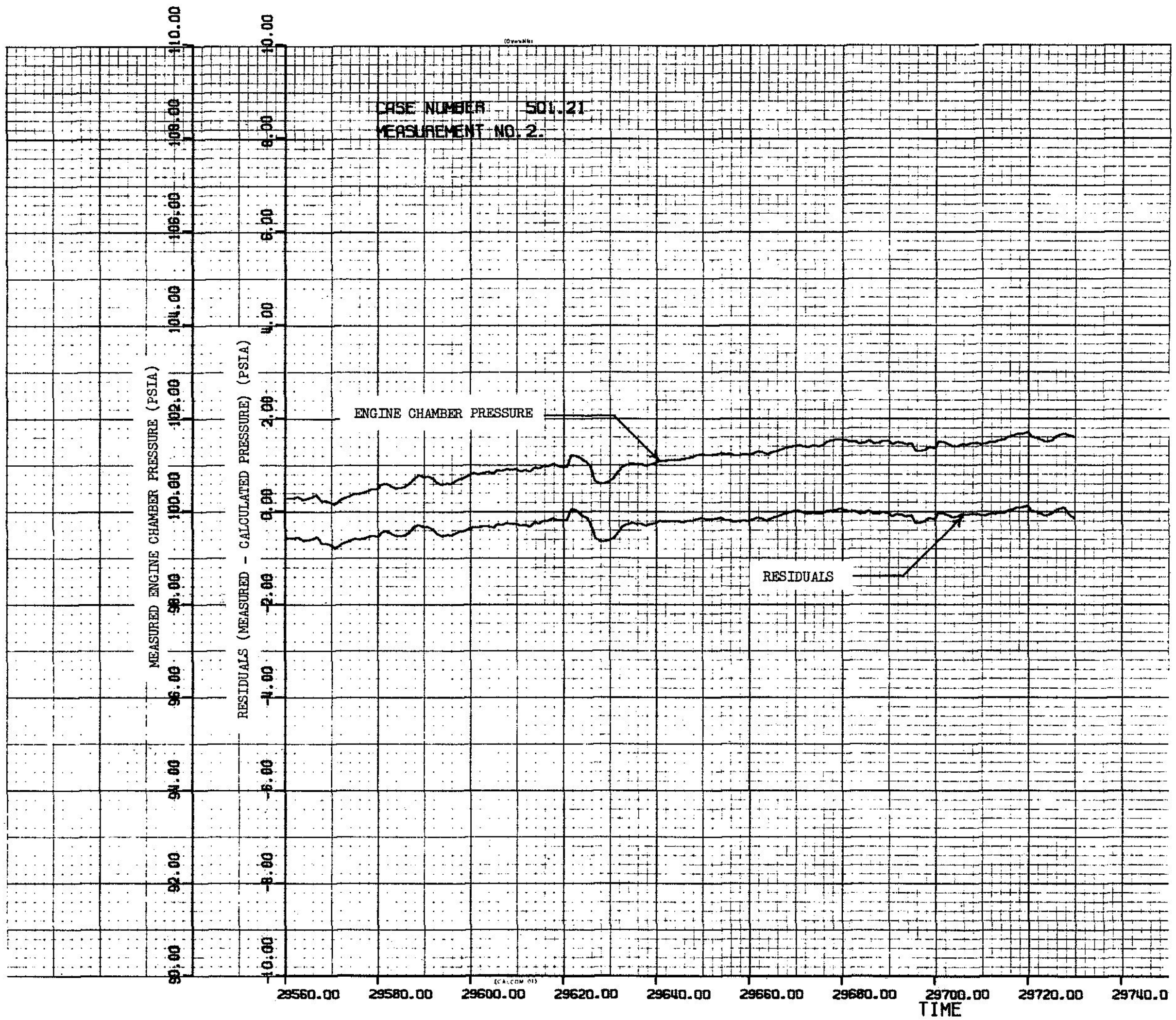



Figure 15. - Engine chamber pressure match — second burn (after crossover).

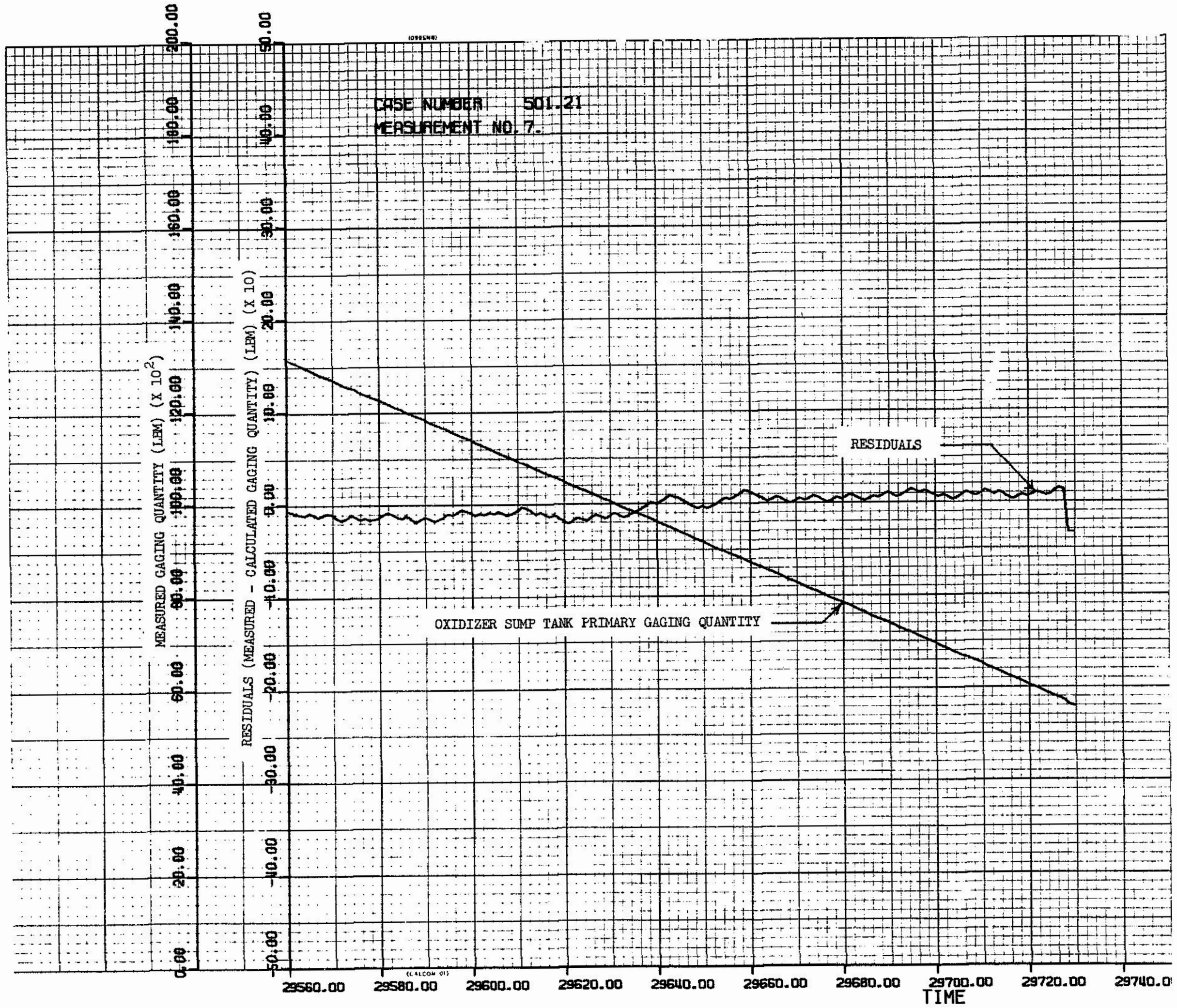



Figure 16. - Oxidizer sump tank primary gaging quantity match — second burn (after crossover).

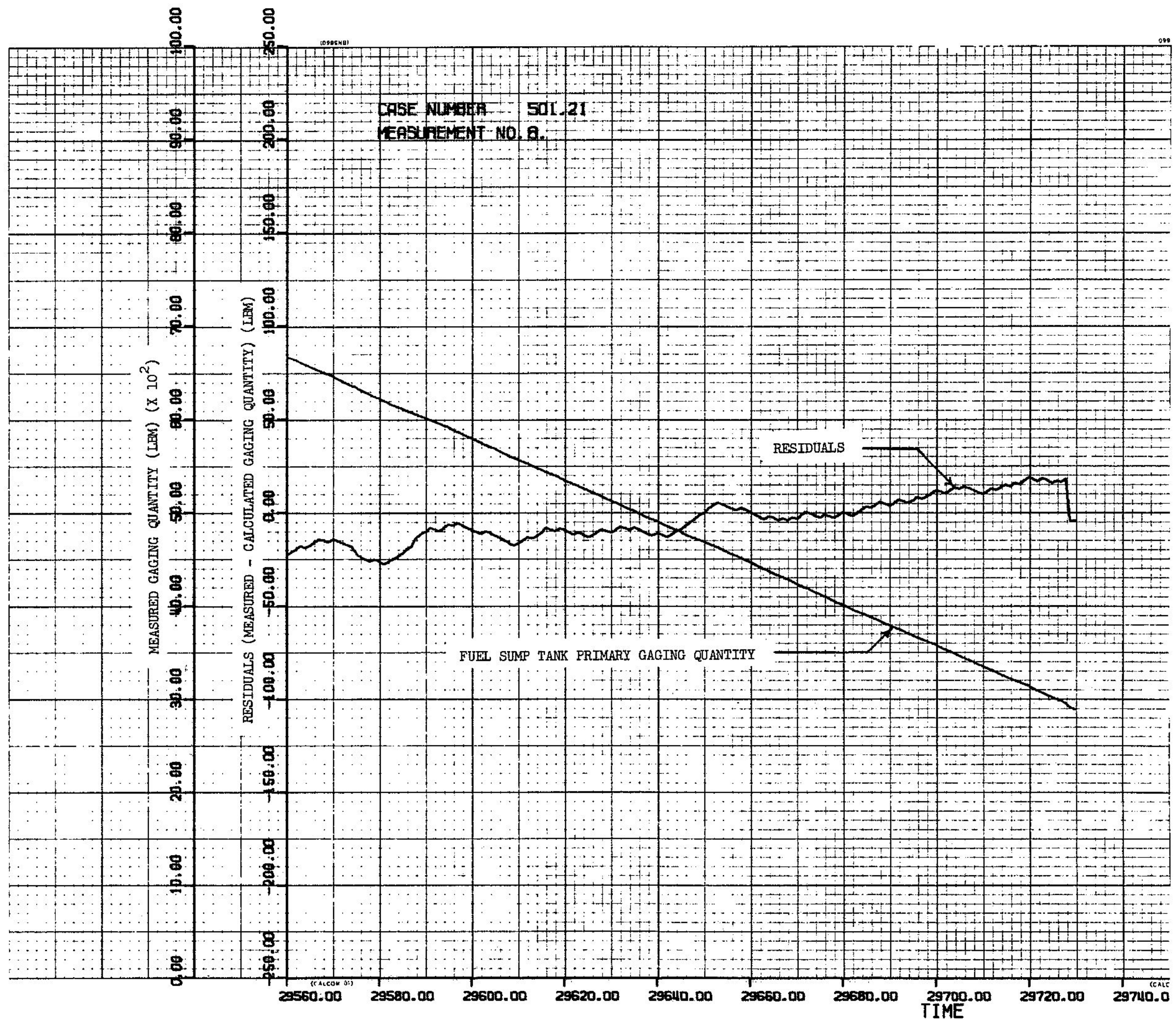



Figure 17. - Fuel sump tank primary gaging quantity match — second burn (after crossover).

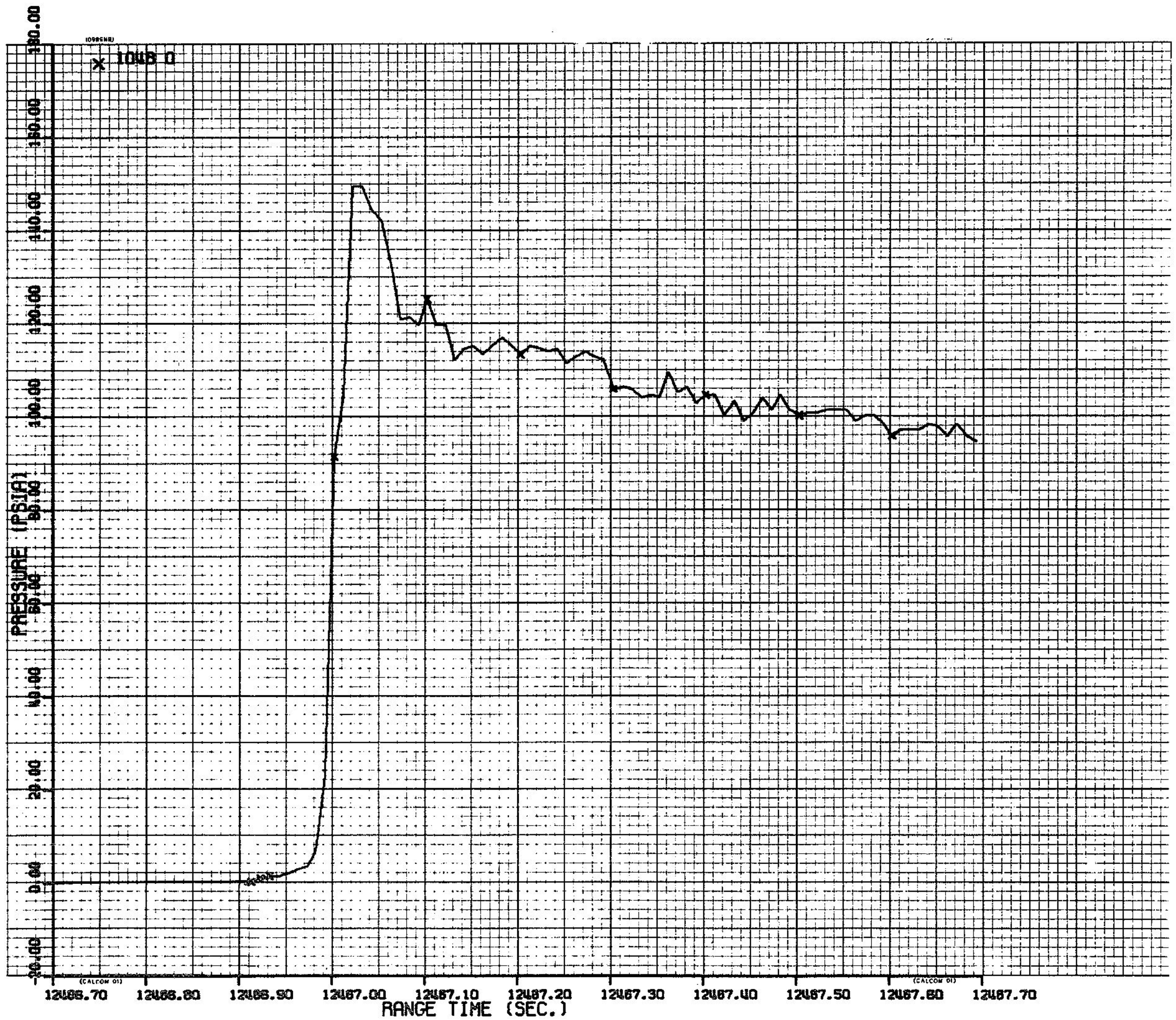



Figure 18. - Engine chamber pressure start transient — first burn.

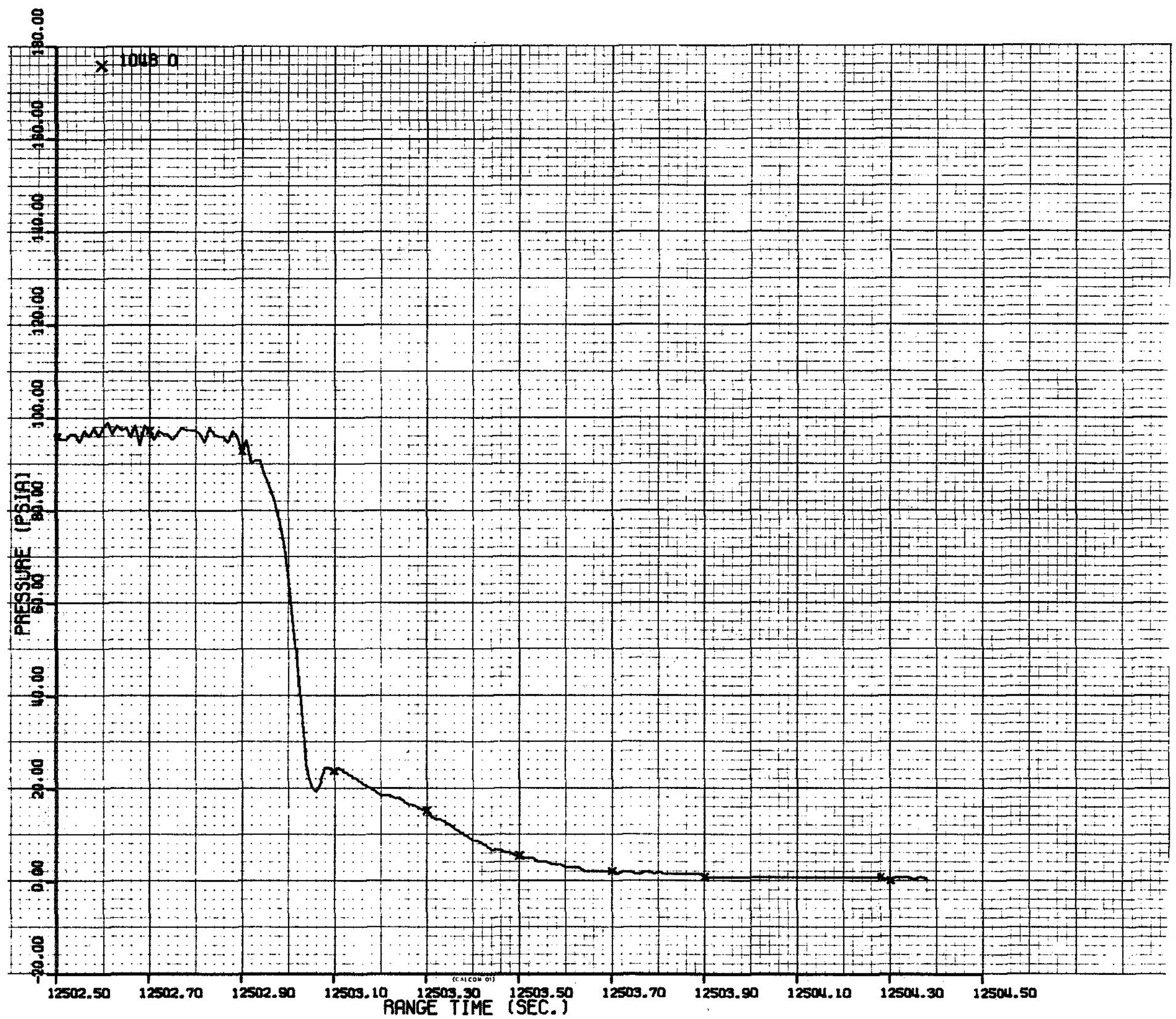



Figure 19. - Engine chamber pressure shutdown transient — first burn.

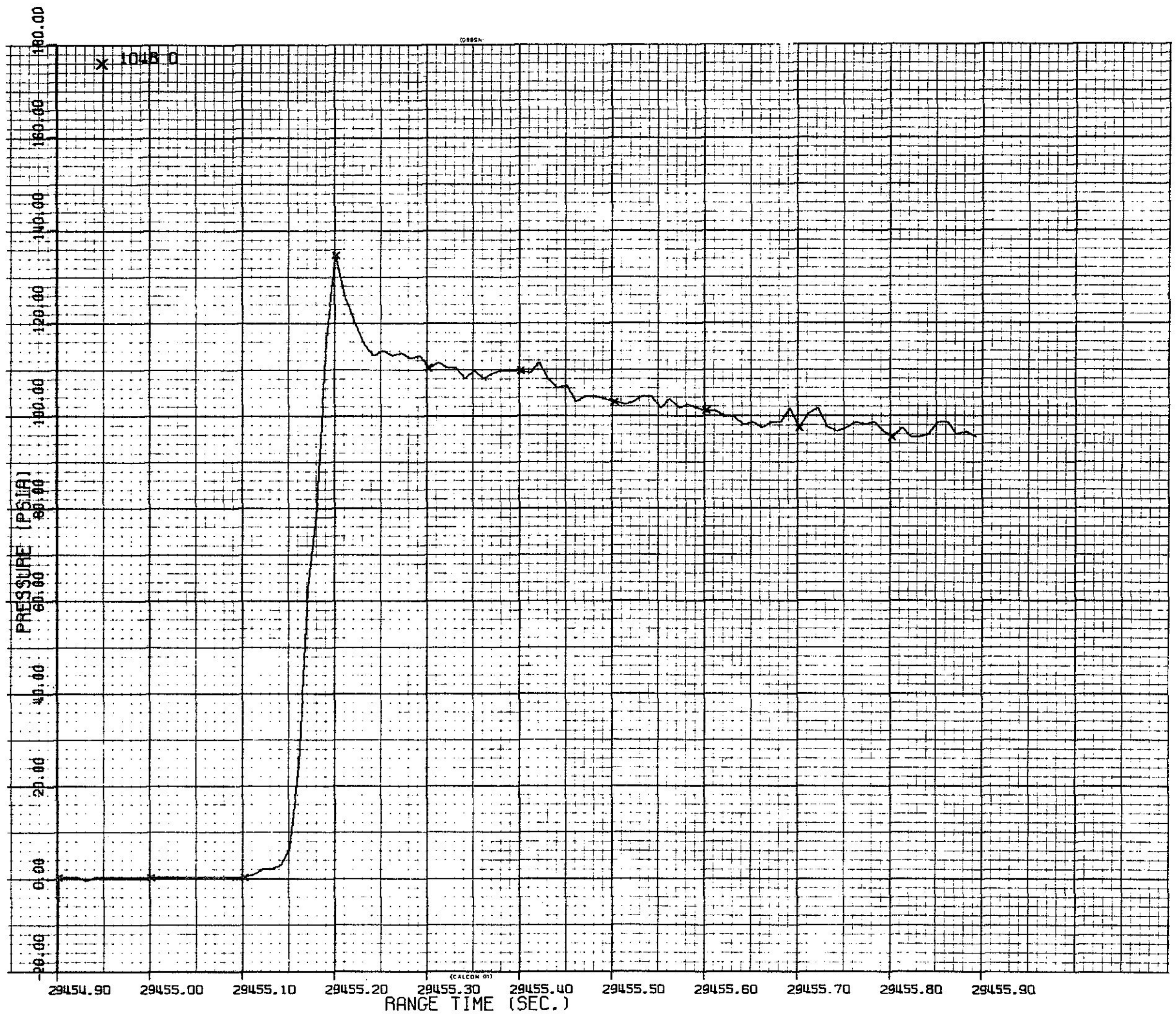



Figure 20. - Engine chamber pressure start transient — second burn.

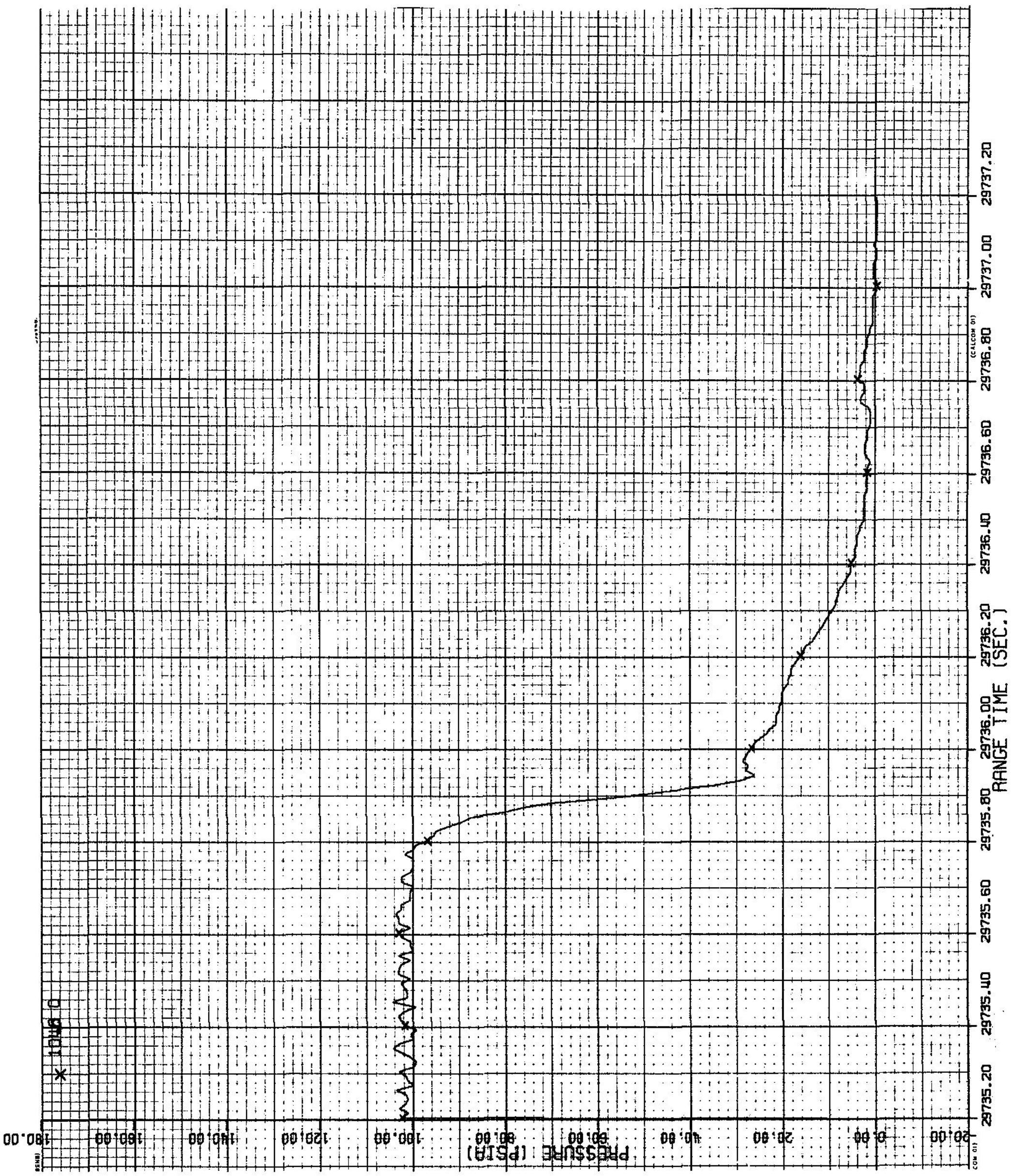



Figure 21. - Engine chamber pressure shutdown transient — second burn.

# APPENDIX

## LISTING OF SPS AS-501 DATA INPUT

## FOR APOLLO PROPULSION FLIGHT ANALYSIS PROGRAM

```
$DIMS
MD1=6,
MD2=16
MD3=2,
MD4=10
$END
$INPT
INSEQ=1,21,30,8,40,26,
KSEQFM(1)=66,67,7,8,9,10,11,12,69,13,14,68,61,15,16,17,18,19,20,21,22,
KSEQFM(30)=37,38,39,40,65,41,42,62,
KSEQFM(40)=55,56,43,44,45,46,47,48,49,50,51,52,53,54,23,24,25,26,27,28,
29,30,31,32,33,34,
INDLOC=1,2,75,76,83,3,9,10,11,12,
LOCTRA=1,2,75,76,83,3,
LOCTRA(7)=11,12,87,88,92,93,121,122,123,124,77
DIAGJ=5625.,900.,25.,25.,676.,10000.,
NUMU=6,
NUMV=4,
NUMEQ=4,
LOCEQ=71,72,73,74,
MAXIT=10,
EPS=.001,
NPRIN(5)=1,
JP=150
IDER(1)=129,130,
YY=15.96,100.,157.,157.,181.,181.,13200.,6727.,0.,0.,
RANGE=20.,RANGE1=20.,
VALUE(9)=46.,23.,
VALUE(13)=17.75,17.75,
VALUE(15)=90.282,56.488,
VALUE(26)=7606.25,
VALUE(42)=89.82,56.3
VALUE(54)=83.95,126.25
VALUE(58)=22.3,60.2,
VALUE(60)=344.16,887.20,
VALUE(81)=0.,1.,
VALUE(100)=6.0,6.0,
VALUE(106)=0.0,0.,
VALUE(110)=-0.1125,-0.1125,
VALUE(114)=36.8,170.4,
VALUE(116)=0.0,0.0,-1.667,-1.667,
VALUE(125)=1.507,1.478,
VALUE(132)=120.,
VALUE(178)=0.,
VALUE(180)=0.36,0.66,4.40,4.96,
VALUE(184)=166.78,133.53,1.065,0.745,-12.96,-13.05,
VALUE(190)=32.174,
VALUE(191)=-1.0,VALUE(193)=1.0,144.0,0.0833333,12.0,2.54145,
SIGMA=.02,5.,3.,3.,3.,3.,160.,80.,480.,240.,
XX=14756.,7357.,0.,0.,0.,0.,
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INDEX(1, 7)=1,42,15,44,183,181,
INDEX(1, 8)=1,43,16,45,182,180,
INDEX(1, 9)=11,1,44,32,
INDEX(1,10)=11,2,45,33,
INDEX(1,11)=14,32,15,23
INDEX(1,12)=14,33,16,24,
INDEX(1,13)=16,131,25,26,500,2,1,
INDEX(1,14)=14,26,25,27,
INDEX(1,15)=5,23,28,50,30,184,186,
INDEX(1,16)=5,24,29,51,31,185,187,
INDEX(1,17)=16,28,52,24,45,2,1,
INDEX(1,18)=16,29,53,27,93,2,1,
INDEX(1,19)=16,30,34,30,147,2,1,
INDEX(1,20)=16,31,35,33,207,2,1
INDEX(1,21)=10,50,58,21,56,188,186
INDEX(1,22)=10,51,59,22,57,189,187
INDEX(1,23) = 3, 69, 9, 54,11,15,110,108,190,193
INDEX(1,24) = 3, 70,10, 55,11,16,111,109,190,193
INDEX(1,25) = 3,108, 9, 13,11,15,106, 67,190,193
INDEX(1,26) = 3,109,10, 14,11,16,107, 68,190,193
INDEX(1,27)=3,67,9,81,11,15,34,46,190,193,
INDEX(1,28)=3,68,10,81,11,16,35,47,190,193,
INDEX(1,29) = 3, 46, 9, 56,11,15, 21,104,190,193
INDEX(1,30) = 3, 47,10, 57,11,16, 22,105,190,193
INDEX(1,31)=3,104,9,81,11,15,52,4,190,193,
INDEX(1,32)=3,105,10,81,11,16,53,5,190,193,
INDEX(1,33) = 13,4,100,92
INDEX(1,34) = 13,5,101,93
INDEX(1,37)=6,9,12,118,112,42,11,60,71,190,
INDEX(1,38)=6,10,12,119,113,43,11,61,72,190,
INDEX(1,39)=7,9,10,62,63,
INDEX(1,40)=15,12,62,27,64,273,442,
INDEX(1,41)=15,12,62,27,65,293,439,
INDEX(1,42)=9,12,82,133,63,25,82,65,81,27,73,66,190,
INDEX(1,43)=18,9,15,11,125,17,197,
INDEX(1,44)=18,10,16,11,126,18,197,
INDEX(1,45)=11,34,17,19,
INDEX(1,46)=11,35,18,20,
INDEX(1,47)=16,19,98,30,313,2,1,
INDEX(1,48)=16,20,99,33,373,2,1,
INDEX(1,49)=17,98,15,121,193,
INDEX(1,50)=17,99,16,122,193,
INDEX(1,51)=17,28,15,123,193,
INDEX(1,52)=17,29,16,124,193,
INDEX(1,53)=3,112,9,114,11,42,116,69,190,193,
INDEX(1,54)=3,113,10,115,11,43,117,70,190,193,
INDEX(1,55)=12,9,129,191,193,
INDEX(1,56)=12,10,130,191,193,
INDEX(1,59)=10,50,137,139,102,178,186,
INDEX(1,60)=10,51,138,140,103,179,187,
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        124.253,12.084,126.971,12.334,129.544,12.584,131.877,12.834,
        133.695,13.061,134.963,13.251,136.168,13.5,136.819,13.834,
TABLE(439)=100.,2.,62.5,
TABLE(442)=3*0.,
TABLE(500)=121.66,0.,121.66,5.,121.62,10.,121.54,20.,121.47,30.,
  121.41,40.,121.34,50.,121.28,60.,121.24,70.,121.17,80.,121.14,90.,
  121.09,100.,121.05,110.,121.0,120.,120.97,130.,120.93,140.,120.90,150.,
  120.86,160.,120.82,170.,120.79,180.,120.75,190.,120.71,200.,
  120.69,210.,120.65,220.,120.63,230.,120.40,300.,
TABLE(560)=21264.5,0.,21264.5,400.,
LIST=9,2,3,5,7,8,
NLIST=1,1,1,1,1,1,
DELPRT=4.0,
MU=10.,
EPT=1.08
MAXTIM=10,
CC=75.,30.,5.,5.,26.,100.,
IK(2)=0,IK(20)=1,IK(24)=1,IK(26)=1,
IK(32)=10,1,1,1,1,
IK(39)=1,1,
IK(45)=104,
INTAPE=11,2,5,6,0,0,8,10,0,0,1,
NWORDS=11,
BURN=170.,
TOPT=2,
ISCAL=1,
YPLOT=1,2,7,8,9,10,
YRMAX=.25,10.,500.,250.,500.,250.,
YRMAX(7)=500.,250.,500.,250.,
YMIN=10.,90.,0.,0.,0.,0.,
YMIN(7)=4*0.,
YMAX=30.,110.,20000.,10000.,10000.,5000.,
YMAX(7)=20000.,10000.,10000.,5000.,
DELMAX=4.0,
$END
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INDEX(1,61)=8,1,2,3,78,80,150,
INDEX(1,62)=4,11,65,12,25,78,89,74,190,
INDEX(1,65)=13,64,83,133,
INDEX(1,66)=13,87,75,112,
INDEX(1,67)=13,88,76,113,
INDEX(1,68)=16,77,150,2,560,2,1,
INDEX(1,69)=13,77,132,131,
TABLE(1)=95.8,-.08,
TABLE(21)=58.56,-.031,
TABLE( 45)=0.,0.,.083,.138,.167,.363,.333,1.054,.5,2.043,.667,3.603,
           .833,4.803,1.,6.512,1.167,8.408,1.333,10.455,1.667,14.893,
           2.084,20.787,2.5,26.746,5.,62.501,6.667,86.338,8.333,110.175,
           10.,134.012,11.418,154.274,11.75,159.034,12.144,164.475,
           12.334,166.955,12.668,170.859,13.084,174.666,13.666,177.206,
TABLE( 93)=0.,0.,.083,.128,.167,.331,.333,.946,.5,1.814,.667,2.907,
           .833,4.196,1.,5.651,1.167,7.243,1.5,10.723,1.75,13.478,
           1.833,14.405,2.084,17.189,2.5,21.828,3.334,31.107,5.,49.666,
           6.667,68.224,8.334,86.783,10.,105.341,11.834,125.755,12.,127.61,
           12.084,128.535,12.418,132.150,12.584,133.843,12.834,136.180,
           13.251,139.266,13.668,140.946,
TABLE(147)=0.,0.,.083,.047,.167,.182,.333,.693,.5,1.502,.667,2.582,
           .833,3.902,1.0,5.432,1.167,7.146,1.333,9.013,1.667,13.091,
           2.084,18.534,2.5,24.043,2.917,29.551,3.334,35.061,4.167,46.078,
           4.75,53.79,4.834,54.95,6.667,81.08,8.334,104.835,10.,128.591,
           11.418,148.783,11.75,153.527,12.084,158.155,12.418,162.448,
           12.773,166.440,12.918,167.769,13.084,169.123,13.418,171.026,
           13.668,171.634,
TABLE(207)=0.,0.,.083,.037,.167,.150,.333,.584,.5,1.271,.667,2.184,
           .833,3.293,1.,4.567,1.167,5.979,1.5,9.097,1.75,11.582,
           1.833,12.419,2.084,14.932,2.5,19.12,2.917,23.309,3.334,27.497,
           3.667,30.848,3.750,31.749,4.167,36.374,5.,45.625,6.667,64.126,
           8.334,82.628,10.,101.13,11.834,121.481,12.,123.331,
           12.084,124.253,12.334,126.971,12.584,129.544,12.834,131.877,
           13.061,133.695,13.251,134.963,13.501,136.168,13.834,136.819,
TABLE(273)=5179.639,6.909,-.02691,0.,492.904,-218.137,
TABLE(293)=1.7524,9.49319E-7,8.9583755E-8,0.,1.76052E-2,-2.00374E-2,
TABLE(299)=0.,-7.4943E-3,9.2371E-3,0.,-3.9620226E-6,-4.41087E-7,
TABLE(305)=1.2359E-4,
TABLE(313)=0.,0.,.047,.083,.182,.167,.693,.333,1.502,.5,2.582,.667,
           3.902,.833,5.432,1.,4.146,1.167,9.013,1.333,13.091,1.667,
           18.534,2.084,24.043,2.5,29.551,2.917,35.061,3.334,46.078,4.167,
           53.793,4.75,54.95,4.834,81.08,6.667,104.835,8.334,128.591,10.,
           148.783,11.418,153.527,11.75,158.155,12.084,162.448,12.418,
           166.44,12.773,167.769,12.918,169.123,13.084,171.026,13.418,
           171.634,13.668,
TABLE(373)=0.,0.,.037,.083,.15,.167,.584,.333,1.271,.5,2.184,.667,
           3.293,.833,4.567,1.,5.979,1.167,9.097,1.5,11.582,1.75,
           12.419,1.833,14.932,2.084,19.12,2.5,23.309,2.917,27.497,3.334,
           30.848,3.667,31.749,3.75,36.374,4.167,45.625,5.,64.126,6.667,
           82.628,8.334,101.13,10.,121.481,11.834,123.331,12.,
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